NASA CR-66509
(NSL 67-300)

ORBITING EXPERIMENT FOR STUDY OF EXTENDED WEIGHTLESSNESS

Volume III

SPACECRAFT PRELIMINARY DESIGN

Prepared under Contract No. NAS1-6971 by
NORTHROP SYSTEMS LABORATORIES
Hawthorne, California
for
Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

This document constitutes a portion of the final report under contract NAS 1-6971, Orbiting Experiment for study of Extended Weightlessness, for the Langley Research Center, National Aeronautics and Space Administration, Hampton, Virginia. The following 6 documents comprise the total report:

NASA CR-66507	Volume I	Summary
NASA CR-66508	Volume II	System Definition
NASA CR-66509	Volume III	Spacecraft Preliminary Design
NASA CR-66510	Volume IV	Laboratory Test Model
NASA CR-66511	Volume V	Program Plans
NASA CR-66512	Volume VI	Orbiting Primate Spacecraft Applications

This report summarizes the results of a definition study of a spacecraft system to support two primates in unattended, weightless, earth-orbital flight for extended periods of time. The experiment is planned as part of the Apollo Applications Program; the spacecraft launched as a LEM substitute on an AAP flight; the primates recovered by Astronaut EVA on a later flight and returned to earth in retrieval canisters within the Command Module. Intensive post-flight examination is planned to ascertain even subtle physiological changes in the primates due to their extended exposure to weightlessness. The study includes definition of mission profile and Apollo Applications Program interfaces, preliminary design of the spacecraft, and planning for subsequent phases of the program.

CONTENTS

ILLUSTRATIONS

Page

Page

APPENDIX A

APPENDIX C

TABLES

Page

LIST OF ABBREVIATIONS

A	Analog
AAP	Apollo Applications Program
ACE	Automatic Checkout Equipment
ACM	Apollo Command Module
ACS	Attitude Control System
A/D	Analog to Digital
AFB	Air Force Base
AGC	Automatic Gain Control
Ag-Zn	Silver-Zinc
APC	Automatic Phase Control
APIC	Apollo Parts Information Center
AM	Airlock Module
ASME	American Society for Mechanical Engineers
ASPR	Armed Services Procurement Regulation
ATM	Apollo Telescope Mount
AVD	Avoidance Component
B	Biological
BCD	Binary Coded Decimal
BLO	Phase Lock Loop Bandwidth of Ground Receiver
BPM	Beats Per Minute
BW	Bandwidth
C	Control (Present)
C&C	Command and Control
CCW	Counter-Clockwise
C/D	Count Down

CDR	Critical Design Review
CEI	Contract End Item
CG	Center of Gravity
CIBA	CIBA Parmaceutical Company
C_L	Centerline
CM	Command Module
Cmds	Commands
C/O	Checkout
CO_2	Carbon Dioxide
CONFAC	Configuration Factor Computer Program
CRB	Configuration Review Board
CSM	Command Service Module
CW	Clockwise
D	Degradation
DAF	Data Acquisition Facilities
DB	Decibel
DCASR	Defense Contract Administrative Service Region Agent
DFO	Director of Flight Operations
DMU	Dual Maneuvering Unit
DOD	Department of Defense
DR	Discrepancy Report
DRD	Document Requirement Description
DRL	Data Requirements List
DRR	Document Request and Release
DSIF	Deep Space Instrumentation Facilities
E	Engineering
E	Event

ECG	Electrocardiogram
ECP	Engineering Change Proposal
ECS	Environmental Control System
ECU	Environmental Control Unit
EDS	Experiment Data System
EKG	Electrocardiogram
EO	Engineering Order
EMI	Electromagnetic Interference
ETR	Eastern Test Range
EVA	Extravehicular Activity
EXC	Exercise Component
FAB	Fabrication
FACI	First Article Configuration Inspection
FARADA	Failure Rate Data Program
FC-75	Minnesota Mining & Manufacturing (Product Designator)
FM	Frequency Modulated
FMEA	Failure Mode, Effect, and Analysis
FMECA	Failure Mode, Effect, and Criticality Analysis
FOV	Field-of-View
FSC	Flight Spacecraft
FTM	Functional Test Model
GAEC	Grumman Aircraft Engineering Corporation
G&C	Guidance and Control
GCPY	Gas Consumed Per Year
Gen.	Generator
GETS	Ground Equipment Test Set

GFE	Government Furnished Equipment
Gnd	Ground
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
h	Unit of Hysteresis
Hg	Mercury
Hz	Hertz
ICD	Interface Control Document
I.D.	Inside Diameter
IDD	Interface Definition Document
IDEP	Interservice Data Exchange Program
ILK	Interlock Task
I/O	Input/Output
IR	Infrared
IU	Instrument Unit
KC	Kilocycles
Kg	Kilogram
KSC	Kennedy Space Center
LEM	Lunar Excursion Module
LES	Launch Escape System
LiOH	Lithium Hydroxide
LM	Lunar Module
LMS	Lunar Mapping System
LMSS	Lunar Mapping and Survey System
LOS	Line-of-Sight
LRC	Langley Research Center

LSS	Life Support System
LTM	Laboratory Test Model
LUT	Launch Umbilical Tower
LV	Launch Vehicle
M	Mission
MAA	Maintenance Assembly Area
MAAS	Manufacturing Assembly and Acceptance Sheet
MC	Control Moment
MCC-H	Mission Control Center - Houston
MCP	Management Control Plan
MDA	Multiple Docking Adapter
MEI	Master End Item
MFOD	Manned Flight Operations Division
MHz	MegaHertz
MLF	Mobile Launch Facility
M&O	Mission and Operations
MOL	Manned Orbiting Laboratory
MRB	Materials Review Board
MS	Multiple Schedule
MSC	Manned Spacecraft Center
MSF	Manned Space Flight
MSFC	Marshall Space Flight Center
MSFN	Manned Space Flight Network
MSOB	Manned Spacecraft Operation Building
M/VM	Mass/Volume Measurement
M/V MD	Mass/Volume Measurement Device

N	Nuisance
NA	Not Applicable
NAA	North American Aviation, Inc.
NAMI	Naval Aerospace Medical Institute
NASCOM	NASA Communications Division
NASCOP	NASA Communications Operating Procedures
NRZC	Non-Return to Zero Change
NSL	Northrop Systems Laboratories
OCP	Operational Checkout Procedures
O.D.	Outside Diameter
OMSF	Office of Manned Spaceflight
OPS	Orbiting Primate Spacecraft
P	Performance (past)
PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
PCU	Pyrotechnic Control Unit
PDR	Preliminary Design Review
PERT	Program Evaluation Review Techniques
PI	Principal Investigation
PIA	Preinstallation Acceptance (test)
PLSS	Portable Life Support System
PM	Phase Modulation
PPM	Parts Per Million
PRINCE	Parts Reliability Information Center
PSC	Primate Spacecraft
PWR	Power
Q	Quick look

QA	Quality Assurance
QC	Quality Control
QM	Qualification Model
QTM	Qualification Test Model
R	Redundant feature
Rad	Irradiation dose unit of measurement
Rad.	Radius
RCS	Reaction Control System
RF	Radio Frequency
RH	Relative Humidity
RMS	Root Mean Square
RTG	Radioisotope Thermoelectric Generator
S	Saftey
SAA	Saturn Apollo Applications
S/AAP	Saturn Apollo Applications Program
S/C	Spacecraft
SCD	Specification Control Drawing
SCN	Specification Change Notice
Seq	Sequence
SGL	Space Ground Link
SIB	Saturn IB
SLA	Spacecraft LEM Adapter
SM	Service Module
SNR	Signal to Noise Ratio
SPS	Service Propulsion System
SRO	Superintendent of Range Operations
STADAN	Space Tracking and Data Acquisition Network

STM	Structural Test Model
TCM	Thermal Control Model
TCS	Thermal Control Subsystem
TE	Time Estimation
TIG	Tungsten Inert Gas
TIM	Timing Task
TLM	Telemeter
TM	Thermal Model
TTM	Thermal Test Model
TWT	Traveling Wave Tube
UCLA	University of California at Los Angeles
USC	University of Southern California
UV	Ultraviolet
VAB	Vertical Assembly Building
VCO	Voltage Controlled Oscillator
VIG	Vigilance Task
VOM	Volt-Ohmmeter
WMS	Waste Management System
WMU	Waste Management Unit
WTR	Western Test Range

INTRODUCTION

The design goal of the Orbiting Primate Spacecraft is to achieve an optimum balance between reliability, performance, compatibility with the SAA mission and hardware constraints and cost. An important factor in achieving this goal was the maximum utilization of proven techniques and equipment consistent with the demands of the above criteria. The latter served as constant guidelines to the study and design effort which resulted in the spacecraft configuration and hardware design of the subsystems set forth in this document. The alternate considerations and rationale associated with selection of the design approach, described below under Subsystems, are contained in the trade studies, which are presented in document NSL 67-309, "Orbiting Experiment for Study of Extended Weightlessness, Subsystem Trade Studies". (ref. 6)

A list of these trade studies follows in table 1.

The approach for systematizing the analyses required in each of the subsystem areas is reflected in the general formatting used in each trade study. In summary this was as follows:

(1) Requirements and constraints

- (a) Problem statement
- (b) Identification of requirements/constraints and their sources
- (c) Quantitative description of requirements and constraints

(2) Alternate approaches

- (a) Identification of feasible approaches to solution of problem
- (b) Description of significant characteristics and parameters of each approach

(3) Comparison of approaches

- (a) Comparison of alternate approaches on the basis of performance, reliability, availability, costs and any other data pertinent to comparative analyses.

(4) Selected approach

- (a) Description of selected approach
- (b) Rationale for selection

TABLE 1. - SUBSYSTEM TRADE STUDIES

*3.2.1.1	Orbiting Primate Spacecraft Tracking Network Optimization
3.2.1.2	Launch Phase Data Transmission
3.2.1.4	Data Handling
3.2.2.1	Television Mechanization
3.2.2.2	Monkey Motion Monitor
3.2.2.3	Biotelemetry Receiving Equipment
3.2.2.4	Remote versus Centralized Signal Conditioning
3.2.2.5	Radiation Dosimeter Instrumentation
3.2.3.1	Centralized versus Remote Control
3.2.3.2	Data Encoding
3.2.4	Power Subsystem
3.2.5.1	Spacecraft Checkout
3.2.6.1	Atmosphere Supply
3.2.6.2	Atmosphere Control
3.2.6.3	Humidity and Temperature Control
3.2.6.4	Carbon Dioxide Control
3.2.6.5	Contaminant Control
3.2.6.6	ECS Thermal Management
3.2.6.7	Waste Management
3.2.7.1	Food Supply and Dispenser
3.2.7.2	Waterer
3.2.7.5	Animal Cage
3.2.7.6	Recovery Capsule
3.2.8	Thermal Control Subsystem
3.2.9.1	Structure, General Arrangement
3.2.9.2	Construction
3.2.9.4	Primate Recovery Capsule Stowage in CM During Re-entry
3.2.9.5	Orbiting Primate Spacecraft Separation Concepts
3.2.9.6	Deployment Devices (Solar Panel)
3.2.9.7	Mass Measurement
3.2.10	Attitude Control

*Refers to numbering system used in reference 6.

REQUIREMENTS

The requirements influencing the configuration of the spacecraft other than the detailed ones affecting the various subsystems, are set forth below.

(1) Provide an orbiting vehicle accommodating two unrestrained primates for the purpose of studying the behavioral and physiological effects of extended weightlessness.

(2) Provide for compatibility between the spacecraft and the Apollo system, and the SAA missions.

(3) Provide for self-sufficiency for a mission period of six months to one year with the latter the design goal.

(4) Provide for recovery of the primates from orbit by EVA as part of an Apollo SAA mission noting that the primates and container are to be stowed in the CM for return to earth.

(5) Provide for acquisition of primate behavioral and physiological data throughout the mission.

(6) Provide protection from radiation and meteoroid particles.

(7) Considerations for adaptation of the spacecraft as part of a rotating system for application of artificial gravity forces.

DESCRIPTION

The Orbiting Primate Spacecraft, described herein, represents a functionally integrated unit designed for use in the SAA program as an LEM companion or LEM substitute payload on manned or unmanned vehicles with a self sustaining capability of six months to one year.

Orbiting Primate Spacecraft Configuration

The functional characteristics, and interrelationships between the various elements of the spacecraft, and between the spacecraft and its interfaces with the launch vehicle, establish both the external and internal configurations. Figures 1 and 2 and drawings 148-40001, and 148-10000, describe the configurations for both the stowed and in orbit modes. The Master End Item Specification Breakdown is shown in figure 3.

External configuration. - Externally the spacecraft is a right circular cylindrical upper section joined to an octagonal lower section. The cylindrical element is welded and contains the pressurized volume. The sides and top of the cylinder form one removable unit, flange mounted at a sealed joint to

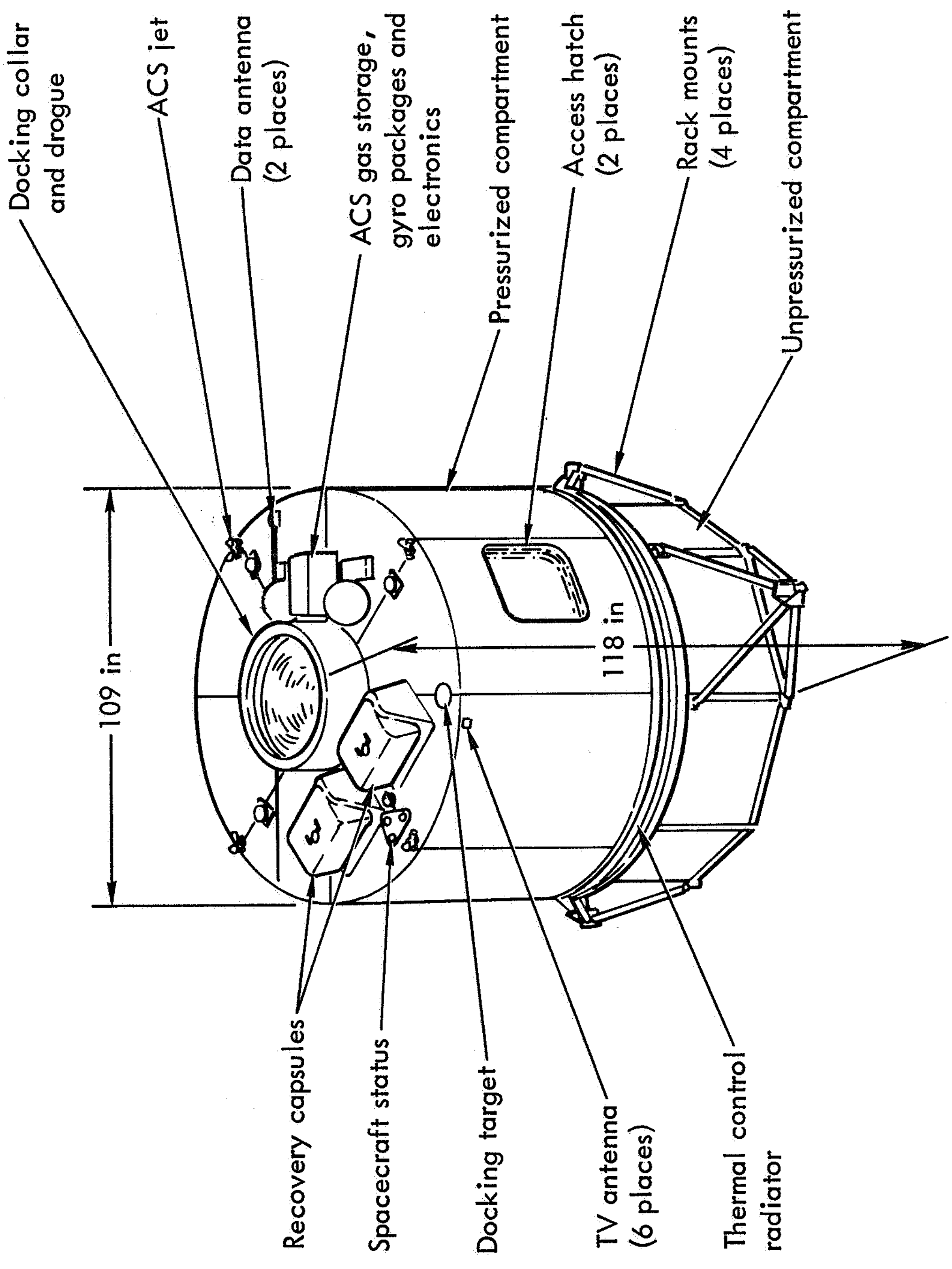


Figure 1. - Spacecraft configuration in stowed position

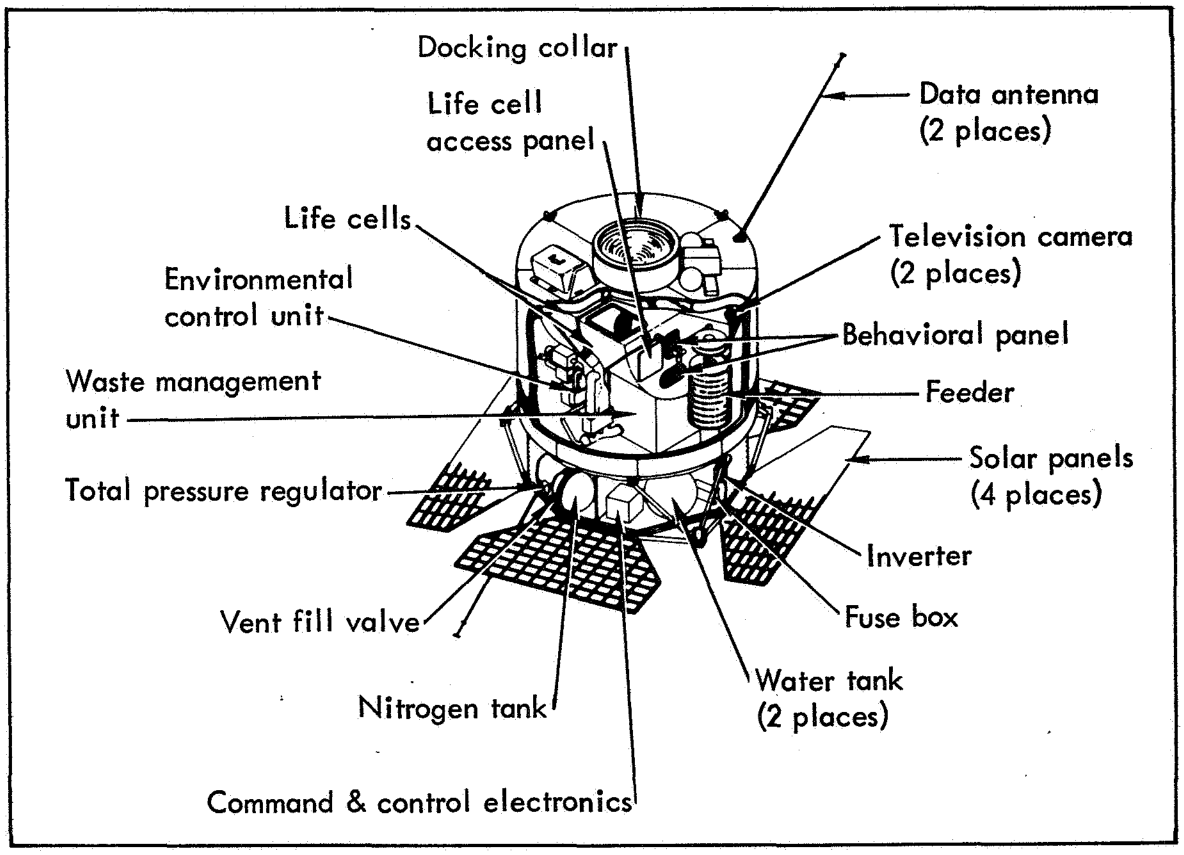


Figure 2. - Spacecraft configuration in deployed position

the bulkhead to form the bottom of the cylindrical section of the spacecraft. The octagonal lower section is unpressurized and contains most of the subsystem equipment. The flat panels forming the sides of the octagonal structure serves as bases and heat sinks for electronic equipment mounted to interior surfaces. A 20 by 20 inch sealable door provides access to the pressurized area for inserting the primate or maintenance of the life cell. Meteoroid shielding panels backed with thermal insulation are attached to the external structural stringers. In addition, the thermal control subsystem radiator is mounted to the external vertical stringer and covers an area of the cylindrical section approximately eleven inches in width and extends completely around the circumference; thereby supplementing the meteoroid shielding and insulation in this region.

Four tubular truss assemblies are utilized to mount the spacecraft to the ATM, LMSS or other appropriate structures in the SLA area of the Apollo Launch Vehicle. These attach points are separated pyrotechnically following docking of the spacecraft and command module. To facilitate docking, a docking collar, located on the upper bulkhead of the spacecraft, contains an LEM type drogue which engages with the command module probe during the docking operations. The primate recovery capsule is also mounted on the upper bulkhead to facilitate removal during extra vehicular activities. In this position, the capsule can be easily reached by the astronaut standing in an open hatch in the command

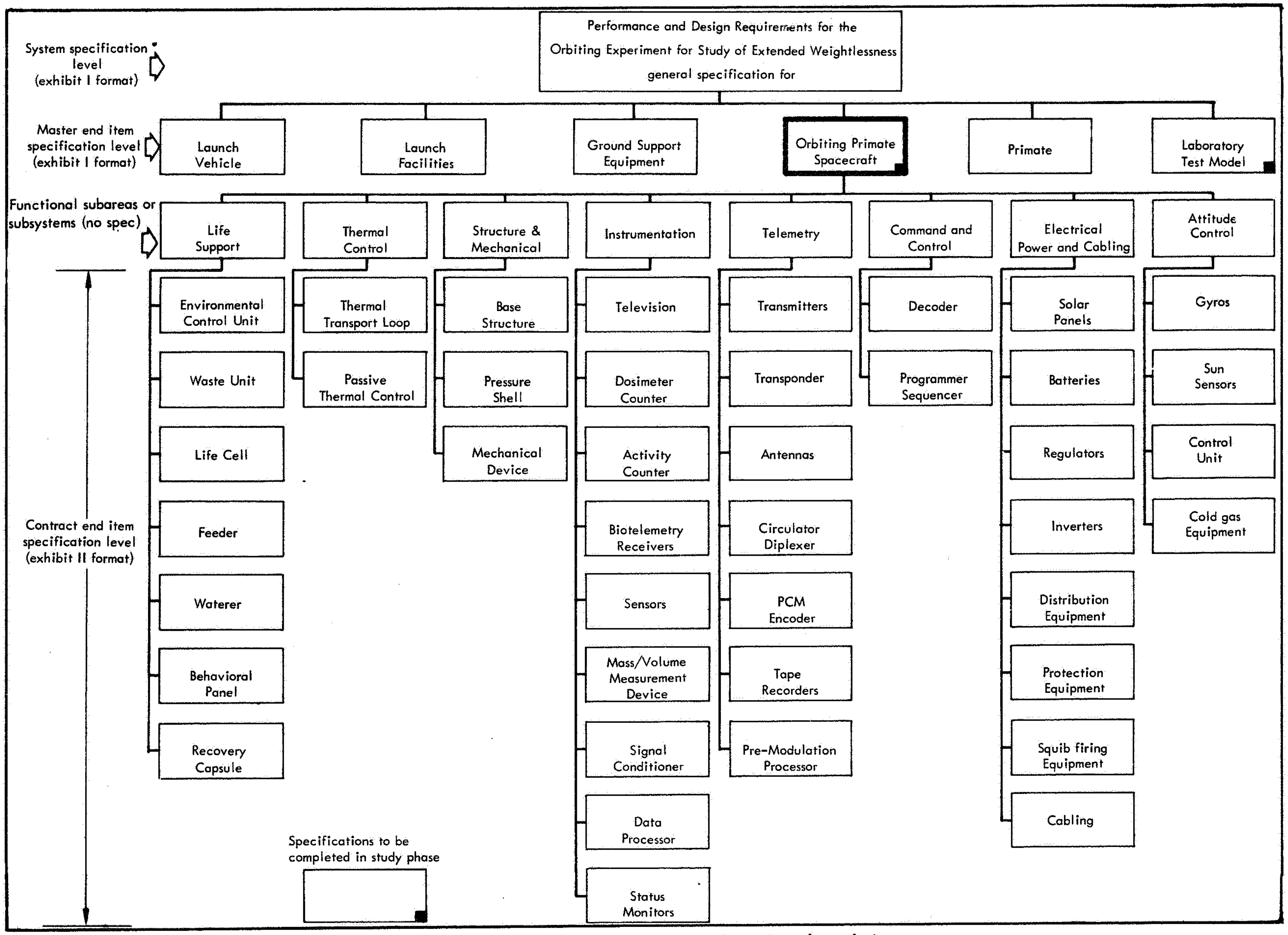


Figure 3. - Master end item specification breakdown

module. In addition, a visual docking aid for use during the docking maneuver and the spacecraft status monitoring panel are visible on the upper bulkhead from the command module window. Within the spacecraft, the upper bulkhead and adjoining area of the cylindrical section provide mountings for the attitude control subsystem tankage, valves, plumbing, and thrusters as well as the gyro module and electronic subassembly.

Five flush-mounted television antennas, four mounted 90° apart around the periphery and one of which is mounted on the top bulkhead are located on the top section of the spacecraft. A boom mounted and deployable communications antenna is also located on the upper bulkhead. The deployed antenna is positioned to eliminate any interference between the spacecraft and the command module during docking. In addition, a television antenna and a communications antenna are mounted at the bottom end of the spacecraft.

The octagonal shaped end of the spacecraft mounts the four paddles which form the solar array. As shown in figure 1, the array panels fold over against the bottom spacecraft surface in the stowed position with their shape conforming with the octagonal outline of the bulkhead to facilitate stowage. Figure 2 depicts the solar array in the deployed condition with the active area facing away from the spacecraft.

Internal configuration. - Two major internal areas comprise the interior of the spacecraft as shown in figure 2: the pressurized volume within the cylindrical section and the unpressurized octagonal section below which contains most of the support subsystems.

The pressurized portion of the spacecraft contains two life cells to accommodate the primates. The two life cells are located side by side with approximately one inch of space between them to permit installation of the structural tension members and to separate the life cell social windows to minimize physical contact between primates. The life cells are mounted to the bottom bulkhead of the pressure vessel. The major equipment attached externally to the life cells include: television cameras, waste management assemblies at the bottom, feeders, waterers and mass measurement devices. The environmental control equipment is also mounted within the pressurized section and interfaces directly with the waste management assembly. The location of environmental control equipment is predicated upon: minimizing the number of openings for conduits, air lines, connectors, etc., from the unpressurized lower section into the pressure vessel; providing a pressurized environment whenever hardware performance and reliability is enhanced in a pressurized environment; and providing requisite proximity between equipments where functional and physical interrelations so dictate. As mentioned earlier, the externally mounted recovery capsules on the upper bulkhead open into the top of the life cells.

Another consideration influencing the location of equipment in the pressure vessel was the desirability of minimizing the equipment mounted on the pressure vessel walls. Since the pressure vessel design permits removal of the upper section, wall mounted equipment requiring electrical or fluid connections or disconnections as part of the removal is incompatible with the design concept and mounting holes, access doors, and so forth would compromise the integrity of the pressure vessel.

Utilizing these criteria as guidelines, only the following major units of environmental control equipment were located in the pressurized area: air filters, condensers, primary, and secondary fans, and a catalytic burner.

Electronic subsystems elements as well as the expendables other than food, which are stored in the Feeder, are located in the unpressurized, lower octagonal section of the spacecraft. The environmental control equipment located in this area consists primarily of: nitrogen and oxygen cryogenic storage tanks, heat exchanger between primary and secondary thermal loops, coolant pump, accumulator, control module, gas analyzer, and lithium hydroxide. The location of the lithium hydroxide unit is also influenced by its need for a thermal input which can best be obtained through the spacecraft surface facing the sun. Therefore, the lower end of the lithium hydroxide container forms part of the solar oriented, bottom surface of the spacecraft.

The expendable items in the environmental control and the spacecraft water are distributed in the spacecraft in such a manner that the weight and balance are optimized during the period of consumption. The location of these items are shown in figure 2 and drawing number 148-10000.

The various equipments are segregated functionally where feasible with consideration given to thermal characteristics, access and maintainability. Thus, the heat dissipating elements of the continuously operating power subsystem are mounted on the exterior flat panels of the octagonal section and serve as heat sinks radiating the heat into space. Conversely, equipment which functions better when it is warm is mounted to the warmer surfaces. The panels, upon which equipment is mounted, are designed to swing out to improve accessibility and maintainability.

Subsystem Summary

The subsystems which comprise the spacecraft have been broken down into the following functional areas:

(1) Life Support
(2) Thermal Control
(3) Structure and Mechanical
(4) Instrumentation
(5) Telemetry
(6) Command and Control
(7) Electric Power and Cabling
(8) Attitude Control

Detailed descriptions of these subsystems are presented in the succeeding sections; however a series of tables are first presented to provide an overview and insight into the functions and interrelationships of the subsystems. An overall block diagram indicating the various relations between subsystems is shown in figure 4.

Life support. - The life support subsystem interfaces directly with the primates, and in doing so performs the functions listed and summarized in table 2.

Thermal control subsystem. - The thermal control subsystem controls the temperature for spacecraft equipment either directly or through a heat sink into which the life support environmental control equipment can reject heat. The various functions and mechanisms of the thermal control subsystem are summarized in table 3.

Structures and mechanical subsystem. - The Structure and Mechanical Subsystem provides the physical support and protection for all the spacecraft equipment and supplies the special mechanizations required by the other subsystems. Table 4 summarizes the functions of the Structure and Mechanical Subsystem.

Instrumentation subsystem. - The Instrumentation Subsystem functions and corresponding mechanizations are summarized in table 5.

Telemetry subsystem. - The Telemetry Subsystem processes and transmits the data from the spacecraft and provides the RF link between the spacecraft and the ground. Its functions and mechanizations are summarized in table 6.

Command and control subsystem. - The Command and Control Subsystem provides preprogrammed timing and sequencing which can be modified by ground commands transmitted uplink on a 2106.4 MHz carrier. Functions and major mechanization of the Command and Control Subsystem are summarized in table 7.

Electric power and cabling subsystem. - The electric power and cabling subsystem generates, regulates conditions, and distributes all of the electric power required by the spacecraft subsystem. These various functions and their mechanizations are summarized in table 8.

Attitude control subsystem. - The attitude control subsystem maintains spacecraft orientation and angular rates within the prescribed limits. The detailed functions and corresponding major mechanizations are summarized in table 9.

Subsystem Interrelationships

The direct interactions and interdependency of the various spacecraft subsystems and the primate are summarized in table 10.

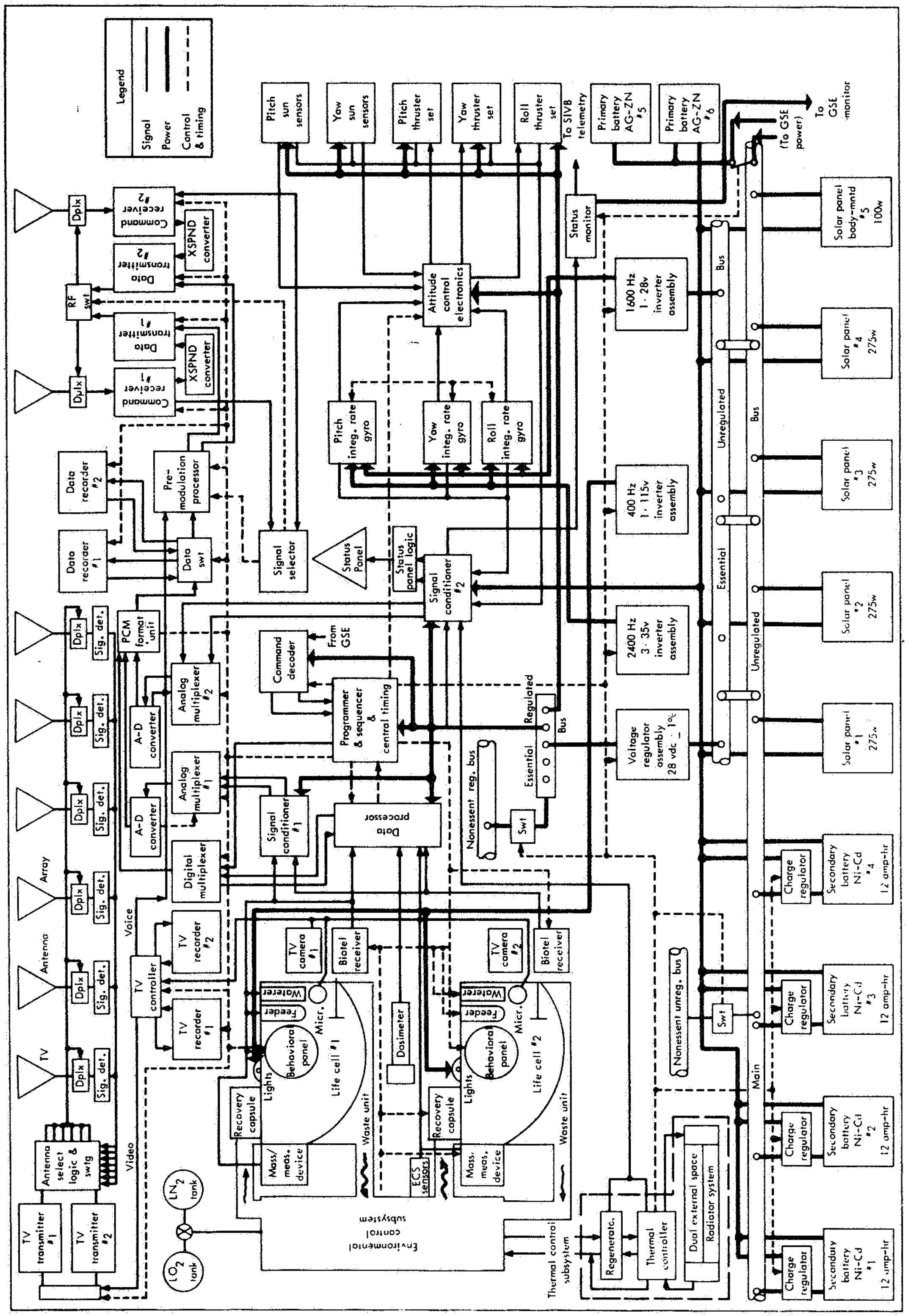


Figure 4. - Spacecraft system block diagram

TABLE 2. - LIFE SUPPORT SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Living area	Approximately 25 cubic feet stainless steel enclosure for primate Lighting installation, TV monitor, exerciser, entry to mass measurement device and recovery capsule provided Contains behavioral panel Contains moving wire mesh wall to assist entry of primate into recovery capsule	The electrified moving wall is the aversive stimulus which the primate escapes (after being shocked) or avoids. There will be four operations per day associated with mass volume measurements as well as assuring entry into the recovery capsule at the end of the mission.
Task performance	Behavioral panel provides visual stimuli and controls for performing tasks, and is located in wall of life cell; food and water supplied through panel	Panel serves as interface between primate and feeder and waterer
Feeding	Food pellets dispensed through mouthpiece in behavioral panel by feeder Food stored in drum in plastic tubing Feeder provides one pellet per task	Food is provided in form of pellets stored in plastic tubing Food is provided as reward for task performed on behavioral panel

TABLE 2. - (continued)

Function	Mechanization	Remarks
Watering	Water supplied through mouthpiece in behavioral panel in 2 cc aliquots in response to lever actuation	Water is provided as reward for task performed on behavioral panel
	Water stored in tanks; forced out by viton bladder compressed by nitrogen	
Environmental control	Oxygen, nitrogen subcritical, cryogenic storage	Two gas atmosphere provided - 21% oxygen approximately 79% nitrogen
	Oxygen partial pressure and atmosphere total pressure monitors - Perkin-Elmer Mass Spectrometer used	Gas flow-function of oxygen partial pressure and atmospheric total pressure
	Triple redundant fans used for circulating atmosphere	Gas enters pressure vessel via regulator and joins atmosphere circulated in pressure vessel cavity
	Condensing heat exchanger for: humidity control, sensible cooling	
	Second heat exchanger for: sensible cooling for varying heat rates	Atmosphere enters life cell at top from pressure vessel acting as plenum; air flows at 30 fpm through life cell and bottom mounted waste units and hence, to circulation, heat exchanging, condensing and CO_2 removal units.
	Heat rejection to heat exchanger coupled to fluid transport loop of thermal control subsystem	
	Lithium hydroxide: removes CO_2; assists in trace contaminant control, bacteria control	Lithium hydroxide high Ph has germicidal affect
	Sorbent beds and catalytic burner provide primary control of trace contaminants	

TABLE 2. - (concluded)

Function	Mechanization	Remarks
	Ultraviolet light used for control of bacteria in air	
Waste management	Scavenging and movement of waste accomplished by mass air flow	Sufficient filter area is provided to assure passage of air over period of mission
	Graded porosity filter used to collect and store waste	
Waste management	Waste is air dried	Waste unit is separated by open mesh from living area
	Waste unit located at bottom of Life Cell	
Recovery	Recovery capsule interfaces with life cell living area	Primary mode of entrance to recovery capsule is via primate training; secondary mode is forced entry by means of moving wire mesh wall, and capsule door
	Outer capsule door provides contaminant cover for inner door which seals capsule	
	Capsule environmental control system is plug-in modular-closed loop recirculating type; oxygen is basic gas; humidity control by desiccant; carbon dioxide control by lithium hydroxide; odor control by use of charcoal; evaporation cooler provides temperature control	Power for environmental control system supplied via spacecraft initially and by umbilical from CM during transfer phase
	Capsule pressure sensed by total pressure regulator, which controls oxygen flow	

TABLE 3. - THERMAL CONTROL SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Provide heat sink for Life Support	Active, transport loop using FC-75 coolant and pumps	Radiator is of fin and tube type
	Interfaces with Life Support heat exchanger and condenser	Regenerator is used to keep coolant from freezing at minimum heat loads
	Space radiator (20 square feet) rejects heat to space	Coolant pumps are triple redundant for reliability
	Radiator inlet and outlet temperatures: Maximum heat load / Minimum heat load in 80°F / -50°F out 55°F / -125°F	Radiator coolant loops are redundant and isolated from each other Coolant supply is adequate to replenish loss of coolant through damaged radiator loop
Provide temperature control for spacecraft equipment	Passive techniques using PV 100 coatings and insulating blankets Judicious matching of equipment characteristics and performance modes with thermal environments	Equipment requiring thermal inputs are mounted to heat sources such as bottom surface of unpressurized equipment bay which receives solar inputs or the bottom of the lower bulkhead of pressure vessel which contains heat source in form on Life Cells. Units which dissipate high levels of heat are mounted to the interior of the flat panels in lower section of the spacecraft. The panel exteriors radiate to space.

TABLE 4. - STRUCTURE AND MECHANICAL SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Provide support and protection for the experiment and sub-system elements	Three hundred cubic foot cylindrical pressure vessel with flat ends. Aluminum alloy construction with stainless steel tension ties between the bulkheads. Upper bulkhead and cylinder sides are removable as a unit for access to the interior. Docking collar on the top of the structure facilitates Apollo command module docking.	Similar, though larger, structure being developed by NASA/MSFC for AAP mission
	Meteroid shielding is provided by a 0.02 2024-T6 bumper to give an annual probability of no penetrations of 0.998	Meteroid analysis supported by measured penetrations in orbit (NASA TN D3717).
	A 150 cubic foot octagonal unpressurized area constructed from welded aluminum alloy extrusions with riveted radial panels	Designed for easy access to the unpressurized equipment
	Attached to ATM rack by a welded truss structure terminated in Apollo LEM Descent Stage attach points	Unpressurized section and lower pressure vessel bulkhead form the base on which to mount and checkout nearly all spacecraft subsystems
Provide separation and deployment from the ATM rack	Four explosive nut/bolt assemblies in an arrangement identical to the Apollo lunar excursion module	Apollo lunar excursion module hardware can be used
Provide docking mechanism	Docking collar and Apollo drogue	Same as Apollo lunar excursion module except for the design of the docking collar below the drogue attach points
Provide for solar panel deployment	Clock-spring powered hinges on each solar panel	Spring powered deployment mechanisms and dampers are same as those used on the Mariner '69 Spacecraft
	Mechanical-hydraulic dampers to absorb deployment shock and latch extended panels	

TABLE 4. - (concluded)

Function	Mechanization	Remarks
	Pyrotechnically actuated	
Provide for antenna erection	Pyrotechnically released antenna	Communication antennas spring loaded and erect against pre-set stops

TABLE 5. - INSTRUMENTATION SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Television monitoring	Apollo camera per life cell	Coverage time equal to daily over-station time
	10° narrow field of vision; 80° wide field of vision	Can be recorded or transmitted in real time
Lighting	Fluorescent, ceiling mounted in life cell	Operates 14 hours per day
	Provides 25 fc at working plane near behavioral panel	
	Controllable	
Dosimetry	Scintillation counter and photo-multiplier	Developed and flown by Northrop on OV1-2 satellite
Bio-instrumentation and activity monitor	Three antennas orthogonally positioned in each life cell wall	Final mechanization will be dependent upon transmitter characteristics

TABLE 5. - (concluded)

Function	Mechanization	Remarks
	Superheterodyne FM/FM receiving and multiplexing	Signal strength variation to indicater motion and activity
	One receiver per antenna	
Mass measurement	Adiabatic measurement of volume in special enclosure by measuring gas pressure changes	Initial tests of technique performed by Northrop; requires additional development to increase accuracy from ±10%
Data sensing	Various available transducers to measure voltage, current, pressure, temperature,	Transducer readily available and used in similar applications
Signal conditioning	Centralized conditioning, plug-in type modules	Concept proved and used in space applications
	Outputs 0-5 vdc	
	Dual isolated outputs per channel	
Spacecraft status monitoring	Multiplexing red line data to SIVB telemetry and prelaunch ground monitor	Locates multiplexer close to data source to reduce box-to-box wiring
	Visual status panel for viewing by astronaut during docking	
Data processing	Behavioral panel stimulus signals	Centralized formating and annotation for various binary-coded data
	Water and feeder monitoring and control	
	Dosimeter signal processing	
	Mass and volume measurement signal sampling and processing	
	Activity monitor signal processing	
Primate voice	Cell mounted microphones, data multiplexed with down-link data	Range of 50 Hz to 12,000 Hz

TABLE 6. - TELEMETRY SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
RF link	Separate transmitter for TV at 2272.5 MHz	Apollo type, equipment is utilized to minimize development
	Six TV antennas; two communication antennas	TV data and voice may be stored or transmitted in real time
	Transponder - transmitting at 2287.5 MHz for data transmission; receiving uplink commands at 2106.4 MHz	Equipment compatible with MSFN
Modulation	Voice - FM - 1.25 MHz subcarrier Data PCM/PM/PM - 1.024 MHz subcarrier	Standard Apollo equipment and frequencies
Data processing	Data encoded and formatted for PCM	Formatting is compatible with MSFN and Apollo data formats PCM approach provides high accuracy
Recording	Separate recorders for data and TV	TV recorder requires development Digital recording of data permits maintaining high accuracy

TABLE 7. - COMMAND AND CONTROL SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Command verification	Command decoder	Verifies that commands are valid Standard Apollo coding
Spacecraft timing and control	Programmer and sequencer	Combination of centralized and remote control Program changes initiated by uplink commands

TABLE 8. - ELECTRIC POWER AND CABLING SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Power generation	Solar array - sun oriented 131 square feet - 1.2 KW - N/P cells	State-of-the-art techniques applicable to solar array
	Primary batteries - AgZn 150 ampere-hours capacity	Primary batteries used in pre-orbital phase of mission approximately five hours
	Secondary batteries - NiCd 48 ampere-hour capacity	Secondary batteries are four in number, diode isolated
Regulation	Unregulated bus, 28 vdc $^{+8}_{-4}$ volts	Unregulated bus operates directly from solar array and battery outputs
	Solar array zener diode voltage limited	
	Buck-boost, 115 W regulator provides 28 vdc ± 0.3 volt	Redundant regulators are used for reliability purposes
Conditioning	Basic dc is conditioned to provide:	Redundant inverters are used for reliability purposes
	20 watts at 35 v, 3 phase 2400 Hz	Each battery has individual charger
	85 watts 115 v, 1 phase 400 Hz	
	1 watts 28 v, 1 phase 1600 Hz	
	Battery charging for secondary batteries	
Distribution	Distribution module provides control point for main power switching	Apollo qualified wire available
Distribution (continued)	Two wire systems utilized	Special shielded pyrotechnic wiring available. Northrop developed and approved
		Twisted shielded pairs of wires used for sensitive circuits
		Segregation of wiring used for electromagnetic interference control
	Wire per MIL-W-16878D	

TABLE 8. - (concluded)

Function	Mechanization	Remarks
distribution (continued)	Fusing employed for circuit protection	
	Over and under voltage circuitry provided	
	Electromagnetic low and high frequency shielded wire used for pyrotechnic circuits	
Pyrotechnics	Squib firing module applies firing power	Squib firing module accepts input low level commands and provides the arming and firing functions using combinations of solid state/relay circuitry
	Separate battery used	
	Circuits isolated from other spacecraft wiring	

TABLE 9. - ATTITUDE CONTROL SUBSYSTEM SUMMARY

Function	Mechanization	Remarks
Establish solar orientation	Utilize sun sensor for initial acquisition and updating	Sun sensors assist in initial acquisition; reference is then maintained by gyros
		Limit cycle is ±10° for angle; ±.002 deg/sec for angular rate
	Cold gas (nitrogen) for propellant	Gas supply is redundant for reliability
	12 thrusters - clusters of 3 each	Subsystem will function with failed jet (open or closed) in any cluster or with loss of complete cluster
	Gas bearing gyros for inertial reference	Capability exists of commanded positioning of spacecraft in other than sun oriented attitude so that during docking CM need not face sun directly in approaching spacecraft docking end
		Gyros are redundant for reliability

TABLE 10. - SUBSYSTEM INTERACTIONS

Performed by \ Performed for	Life support	Thermal control	Structural and mechanical	Instrumentation	Telemetry	Command and control	Electrical power and cabling	Attitude control	Primate
Life support									Provide food, water, shelter, atmosphere and temperature control, exercising, programmed tasks
Thermal control	Provides heat sink using heat exchanger and active loop			Provides temperature control passively, PV 100 coat on mounting panels	Provides temp. control passively (cat-a-lac black coating; insulating blankets)	Provides temp control passively, (cat-a-lac black coating, mtg. on solar heated structure.)	Provides temp control passively, PV 100 coat on mounting panels	Provides temp. control passively insulating	
Structural and mechanical	Provides sealed pressure vessel with 0.2 lb/day leakage structural support, meteoroid protection	Provides structure support, heat sinks meteoroid shielding		Provides structure support, meteoroid shielding, accessibility	Provides structure support, meteoroid shielding, accessibility, antenna stowage and deployment	Provides structure support, meteoroid shielding, accessibility	Provides structure support, meteoroid shielding, accessibility, solar array stowage and deployment	Provides structure support	Provides meteoroid and radiation shielding
Instrumentation	Provides for monitoring 69 parameters, behavioral panel stimulus signals	Provides for monitoring four parameters	Provides for monitoring nine parameters	Provides for monitoring two parameters	Provides for monitoring 11 parameters	Provides for monitoring six parameters	Provides for monitoring 30 parameters	Provides for monitoring 15 parameters	Provides: antennas, receivers, multiplexer for ECG, temp. respiration implant transmitted data; voice, TV monitoring; radiation monitor; acitivity monitoring lighting
Telemetry				Provides RF link for TV and data, encoding and formating conditioned data, records TV and data		Provides receiving and demodulation of command signals			
Command and control Electric power and cabling	Provides programmed and real time commands Provides 115v, single phase, 400 Hz; 28 vdc; wiring	Provides programmed and real time commands Provides 115v, single phase 400 Hz; 28 vdc, wiring	Provides pyrotechnics control and power for separation and deployment devices; wiring	Provides programmed and real time commands Provides 28 vcd; wiring	Provides programmed and real time commands Provides 28 vdc; wiring	Provides 28 vdc; wiring	Provides programmed and real time commands \\ Provides 28 vdc; wiring	Provides programmed and real time commands Provides 28 vdc; 35v; 2400 Hz 3ϕ; 28v, 1600 Hz 1ϕ; wiring	
Attitude control		Provides sun orientation permitting more efficient space radiator design					Provides: solar array orientation to ±10° of sunline		Provides: approx. zero g environment by control of spacecraft rates

Weight Summary

The total spacecraft weight and the estimated individual subsystem weights are presented in table 11. Reflected in the listed weights are redundancies included in certain subsystems to increase reliability of gases, etc. Equipment description lists 148-10001 of reference 8, provide detail component weights. The estimated allowable payload weight for an SAA manned launch is 8,800 pounds, however, for an unmanned launch this weight is 36,000 pounds.

SUBSYSTEMS

The subsystem described in the following sections represent the configurations selected from the results of various analyses and trade studies undertaken in the course of this effort. As such these subsystems reflect presently known requirements and constraints, and are in sufficient depth to assure feasibility and an advanced starting point design in the subsequent phase of the program.

TABLE 11. - WEIGHT SUMMARY

Subsystem	Weight	
Life support	(lb)*	(lb)**
Environmental control and gas storage	965	984
Waste management	92	92
Life cells	320	320
Feeders	340	340
Waterer and storage	700	700
Recovery capsules	105	105
Thermal control	170	176
Structure and mechanical	1209	1511
Instrumentation	159	159
Telemetry	204	292
Command and control	30	30
Electric power and cabling	510	527
Attitude control	82	150
TOTAL	4886	5386

*Redundancy not included; limit pressure safety factor of 1.5

**Includes critical component redundancy and limit pressure structural safety factor of 2

Life Support

The primary requirement of Life Support is to provide an environment that will support and sustain two unrestrained, unattended primates for a period of one year. To satisfy this requirement, the Life Support subsystem must provide the following functional elements: life cell, feeder, waterer, behavioral panel, environmental control/waste management and recovery capsule. The functional relationship of these elements is schematically illustrated in figure 5.

Life cell. - The life cell is the central element of the Life Support Subsystem and serves as an enclosure and living area for the primate. To provide adequate living area for the primate, the suggested dimensions of the life cell are 36 inches high, 30 inches wide and 40 inches long, with an enclosed volume of 25 cubic feet. The life cell interior surfaces must be designed to minimize interferences with television-viewing and inhibit the monkey from grasping the ceiling or orienting himself to the ceiling. The floor must provide footholds for the monkey and present a minimum obstacle to the passage of primate waste products. The cage must contain no components or fasteners that are accessible for removal by the primate. The interior must have a minimum of projections or obstacles which might collect waste products.

The cage materials must resist scratching and chewing by the monkeys, and chemical attack from fresh or decomposing food, feces, and urine.

The cage should be illuminated with an intensity of 25-foot-candles for fourteen hours per day, with a spectrum approximately that of sunlight. During the ten-hour dark period, the light intensity should be 0.01 to 0.10 foot-candle. Provision should be made to reprogram the light source to change either the length of the light period or dark period. The cages should permit social interaction between monkeys to the extent of finger or toe touching. Some form of exercise may be desirable to keep the monkey in good physiological condition, and provision for a monkey actuated exercise device should be considered.

Life cell description and performance: The external configuration of the life cell is represented in figure 6. Externally removable hatches 20 inches by 20 inches are located in the spacecraft pressure shell wall, and a 16 inch by 16 inch access panel is located on the sidewall of each cage. These access openings will permit insertion of the monkeys into the life cell cages late in the launch sequence without the necessity of breaking the recovery-capsule-to-spacecraft interface seals.

These panels will also provide access to the inside of the pressure shell for repairs and adjustments while on the launch pad. Two of the biotelemetry pickup antennas are located in these cage access panels, and the third antenna is in the end of the cage below the mass measurement device.

The life cell cage, shown in figure 6 and drawing 148-11300 ref. 8, is 29.5 inches wide and 36 inches high based on the curved bottom radius of 36 inches with an overall diagonal length of 67 inches. This configuration provides approximately 25 cubic feet of cage volume and a floor area of approximately 13 square feet. This floor area will provide a large surface for the animal to grip and orient himself properly. It also will provide a cross sectional area large enough so that any debris buildup on the grill bars will not affect the air flow normal to it.

The walls and ceiling of the cage are smooth matt finished, stainless steel panels with rounded edges and fitted corners to minimize the collection of debris and the possibility of injury to the animal. The grill floor,

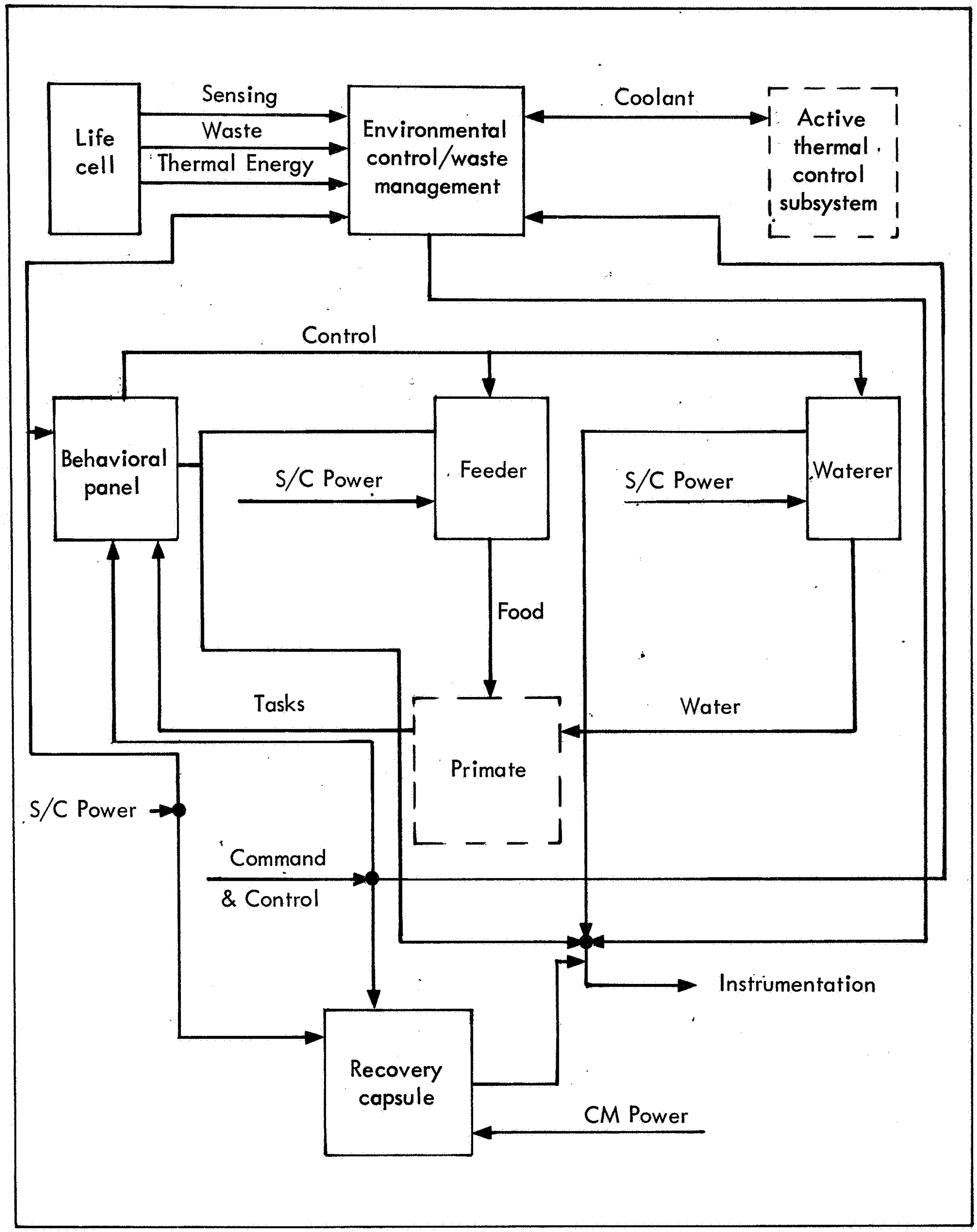


Figure 5. - Life support subsystem block diagram

moving wall, and social opening mesh, are constructed of stainless steel tubes welded together at each intersection.

A one-inch space between the two cages separates the two walls to provide space for wall reinforcements, a pressure shell tie member and cabling runs. This one-inch cage spacing also serves to separate the social opening screens. The selection of one-half inch square grill openings enables the animals to touch each others fingers or toes but prevents them from biting each other through this social opening. Both the social opening and the activity panels are located near the spacecraft centerline to minimize the effect of animal movements on the spacecraft attitude. Conversely, the angular motion of the spacecraft will have the least effect on the animals with this configuration layout.

Welding, riveting, and fasteners unsusceptible to tampering by the monkey will be used throughout the interior construction of the cage and capsule. Corrosion, erosion, and bite and scratchproof materials and finishes, will be used on all surfaces exposed to the primates. Teflon, for example, is too soft to be used on surfaces exposed to the monkey but is suitable for other surfaces where special protection or reduced friction is desired.

The floor grill openings are three-fourths inch wide which prevents a monkey from reaching the waste collector which is at least seven inches away from the grill. The waste collector is a passive system and is described in further detail in succeeding sections. Its location at the bottom of the life cell provides efficient operation in both a gravitational field and under weightless conditions.

The recovery capsule and the mass measurement devices are located at the end of the cage opposite the television camera. This arrangement enables the camera to view these stations with either a wide angle or a narrow angle lens for close surveillance of the animal and the mechanisms.

As detailed in the Trade Studies (ref. 6), both life cell cages are located near the top of the spacecraft to provide expedient and safe recovery operations. However, this cage arrangement makes it difficult to mount the television camera above the cage. In any event, this would be a poor position since it would require a large mesh or grill ceiling for viewing, and would give the animal something to grasp, and thus orient himself improperly. Further, with the television camera above the cage, the view of the monkey and cage is generally degraded due to lens angle limitations, poor overhead view if the primate is oriented in the desirable positions, and poor animal-to-background contrast with the floor grill and waste collector.

The moving wall when in its normal position acts as one end of the cage as shown in figure 6. The wall is constructed of parallel bars on 2-3/4 inch centers, running the full length of the wall. The lower half of the wall has 3/16 inch diameter rods spaced on one-inch centers mounted transversely to the bars. The wide spacing of the upper bars provides an excellent view for the television system and minimum light attenuation. The primate can reach through these openings, up to his armpit ovaxilla but cannot reach the television

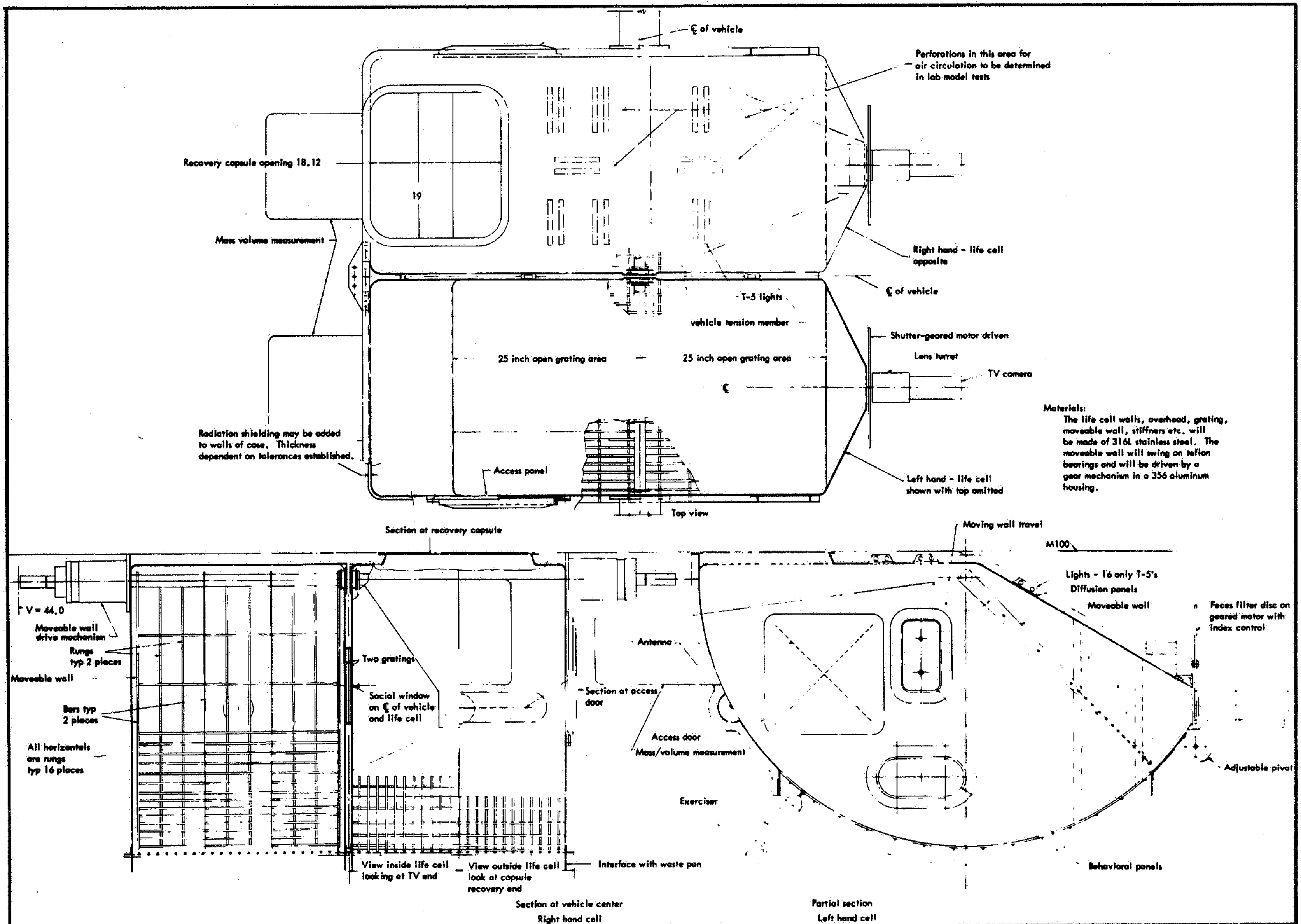


Figure 6. - Life cell

opening which is 17 inches away. The lower portion of the moving wall is less than 17 inches from the television lens but its 3/4 inch grill openings prevent his reaching full arm length.

The moving wall was selected as a positive means of inserting a deceased or a live monkey into the recovery capsule. However, it can perform other secondary functions if desired, such as scraping large pieces of debris off the grilled cage floor and off the walls. These secondary functions could be performed on a planned schedule based on laboratory test experience, or they could be performed at the direction of ground command based on television information. Actual movement of the wall can be accomplished in steps using television checks before each step, or it could be a slow steady motion from start to finish. Consideration has been given to imposing a voltage potential (50 volts at 400 Hz; current limited to 1.5 milliamps maximum) on the moving wall to provide a shock stimulus to the monkey for training or control purposes.

The electromechanical actuator to drive the moving wall is shown located at the upper pivot axis outside the cage. It may, however, be a powered "sheep's foot" roller mounted at the lower edge of the wall this roller would drive the cage similar to a rack and pinion gear by meshing with the floor grill bars and thus provide a cleaning effect of the floor openings as the roller's teeth meshed with the grill.

Advanced development areas: Both the electromechanical actuator and the sheep's foot roller concept have merit. But, in order to select the best and most reliable components and systems, the following areas related to the moving wall drive system should be investigated and tested.

(1) Pivot mounted actuator as shown in figure 6 or drawing number 148-11300 (ref. 8).

(2) Sheep's foot drive roller mounted at floor line to move wall and push debris through floor grill.

To evaluate the pivot mounted actuator, a simple test should be set up using a curved bottom cage and possibly a simple hand-powered actuator to move the wall. Forces and power required to cut through hard and soft feces should be measured and the scraping effect, smear and pile-ups should be checked and photo recorded. The tendencies of the wall to pinch the monkey, and his reaction to the wall should also be noted. The need or desirability of electrifying the wall for shock stimulus can also be investigated at this time. Safety devices can be incorporated into the actuator drive unit to limit the drive torque and prevent injury to the monkey should he get caught or pinched in these tests. One such safety device would be a spring loaded drive system which enables the monkey to override the wall motion for a few degrees. Or, it could incorporate a load sensor which would automatically back the wall off a few degrees and then proceed forward again after sensing a minimum resistance force. Another safety device could be integrated into the wall to return it to the normal position and lock it there should the electric drive actuator malfunction. If this should occur, the wall would not interfere with the normal activities. Of course, the wall could not be used to insert the monkey into the capsule if such a malfunction occurred.

After the pivot actuator tests have been completed, the test set-up should then be modified to include the "sheep's foot" drive roll and the same functions tested as before. It may be necessary to enlarge the cage floor grill length to give the full angular movement to the wall. The effect of this larger floor grill extending to the end of the cage would be determined at this time. With the "sheep's foot" drive system, the collection of waste matter around the roller and again the possibility of catching the monkey would be carefully evaluated.

The position of the lights for uniform daylight illumination, good television viewing, minimum power, acceptable shadowing by grill bars and lens contamination should be determined in a simple test cage. It would be desirable to test several light lens, cleaning devices or techniques. These devices would include redundant fixtures with a retractable cover to prevent contamination when not in use and rotating drum lenses to expose clear areas or scrape off debris. Other devices to scrape or attempt to clean the lenses should also be investigated. The need for these devices or the best selection, if needed, cannot be confirmed until suitable tests have been conducted.

During the tests, it would also be desirable to investigate several promising methods of protecting the television lens from feces and urine. These techniques include a transparent door, shown in figure 6, a roll-up plastic tape, an air curtain (during television operation), and mechanical lens cover surfaces.

In addition to the moving wall and illumination investigations, tests need to be performed to ascertain atmosphere flow patterns within the life cell. Cleaning the life cell is predicated upon the proper flow of dry air, and additional information is needed on the flow velocity, which determines the effective temperature for a given dry bulb temperature, around the primate.

Data is also required on the type of exerciser required to maintain the physiological condition of the primate. One promising exerciser incorporates a swinging handle which is geared to a motor/generator at the handle pivot. This drive unit can be powered to rotate the handle from its stored position parallel to sloped cage ceiling, out through the moving wall grill to the desired operational position. From here the monkey can push or pull the handles as desired through a 30 inch stroke. The force required to push the handle can be controlled from the ground by regulation of the generator's field strength or output power. When the exercise task is finished the generator is again powered to act as a motor and retract the handle to the stored position. Should this device stick in any operating position, it will not interfere with the moving wall operation as the handles and arms are designed to clear the moving wall's grill bars. Another configuration for the exerciser is shown in figure 6 and drawing 148-11300 (ref. 8). The monkey pulls this device straight up from the floor. It also has a 30 inch stroke and could include a variable pull or push force selection. This tire pump type exercise could employ mechanical springs or pneumatics to control the force and retract the handle at the end of the task.

The requirements and constraints of the exerciser should be studied in detail and tests performed to select the best configuration and location.

Preliminary equipment list: A list of major equipment items, other than structural, are itemized in table 12. The equipment evaluated is developed and available.

TABLE 12. - LIFE CELL PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	Geared motor	Globe Industries, Inc.	102A190-10	2
2	Geared motor	Globe Industries, Inc.	102A165-10	2
3	Industrial gear reducer	Harmonic Drive Division, United Shoe Machine Corporation	HDUC 40-120-1-GP	2

Feeder. - The basic animal nutritional requirements are furnished from a storage and dispensing device, either as a reward for activities performed, or ad libitum. The alternative methods of feeding the animals include the food formulation and consistency and mechanization concepts. For instance, the food could be liquid, slurry, gel, granular, or pelletized forms. However, regardless of the form of the food, the primary objectives include reliability, prevention of contamination, and animal acceptance.

With the possible exception of gases, water or aqueous solutions are the easiest of commodities to dispense. Suspensions are next and the solids are the most difficult. However, storage problems and nutritional values are another story and liquid nutrients must be stored at lower temperatures. In addition, at present time, only dry nutrient food has been proved acceptable. A gel food has possibilities, but here again the nutrient and preservation properties have not been proved. The different responses of gel type materials to pressure and temperature variations must be explored further before consideration can be given to such systems.

In view of the above considerations, the feeder was designed for pellitized food. Nutritionally, the animal food will be of a composition tested for laboratory maintenance of Rhesus colonies, such as CIBA Whole Diet Nutrient Pellets. On demand and per program, the monkey will be fed 150 pellets per day. The pellets must be enclosed at all times to prevent contamination. The delivery rate is on an average of one pellet per two and one-half minutes. Upon illumination of the stimulus light, the monkey will place his mouth over the mouthpiece and will receive a pellet after one-half second. For each animal, 54,750 one-gram pellets, weighing 120 pounds must be loaded in the spacecraft in a sterilized condition. Reliability and freedom from contamination are the most important considerations. The primary requirements of the feeder are summarized as follows:

(1) Operates in a one atmosphere and under 1 g conditions during training and prelaunch conditions.

(2) Operates in a one atmosphere under zero g conditions for one year while in orbit.

(3) Dispenses pellets directly into the animal's mouth in a manner acceptable to the animal.

(4) Impervious to jamming by wet or hardened feces, or food in delivery nozzle.

(5) Signals the pellet has been delivered within 0.01 second.

(6) Prevention of contamination of food from sources, such as moisture, feces, urine, or wet food in or on the delivery nozzle.

The preliminary design of the feeder is shown in drawing 148-11400, figure 7. In this design, the pellets are stored by encasing them in a continuous, drum wound, sealed tube. The tube is closed by a zipper and opened by a three pin opening device which opens the tube before the tube enters the capstan. In the flat pattern, the tube is reinforced along both edges which eliminates the hazard of crosswise tearing while the tube is being opened.

To prevent unspooling, the drum is restrained by a spring loaded brake. The braking force is set to slightly exceed the torque produced by the stiffness of the pellet tube with either a full or empty drum, whichever is greater. This means that the capstan pulling the tube must overcome this variable difference. The braking surfaces are non-galling 7075-76 aluminum disks, spring loaded with screw adjustment, which will be wired in final position.

The loaded tube is pulled off the drum over a large guided and shrouded pulley. Since placing this first pull-off lead pulley far enough from the drum to utilize a stationary pull-off is not practical, it is mounted on a swing block and slide, and is motor-driven back and forth to align the pull-off force approximately normal to the drum axis. The total operation of this slide involves traveling back and forth a distance of 28 inches, 14 times, as the drum rotates, and hence reflects a potentially low wear-out factor.

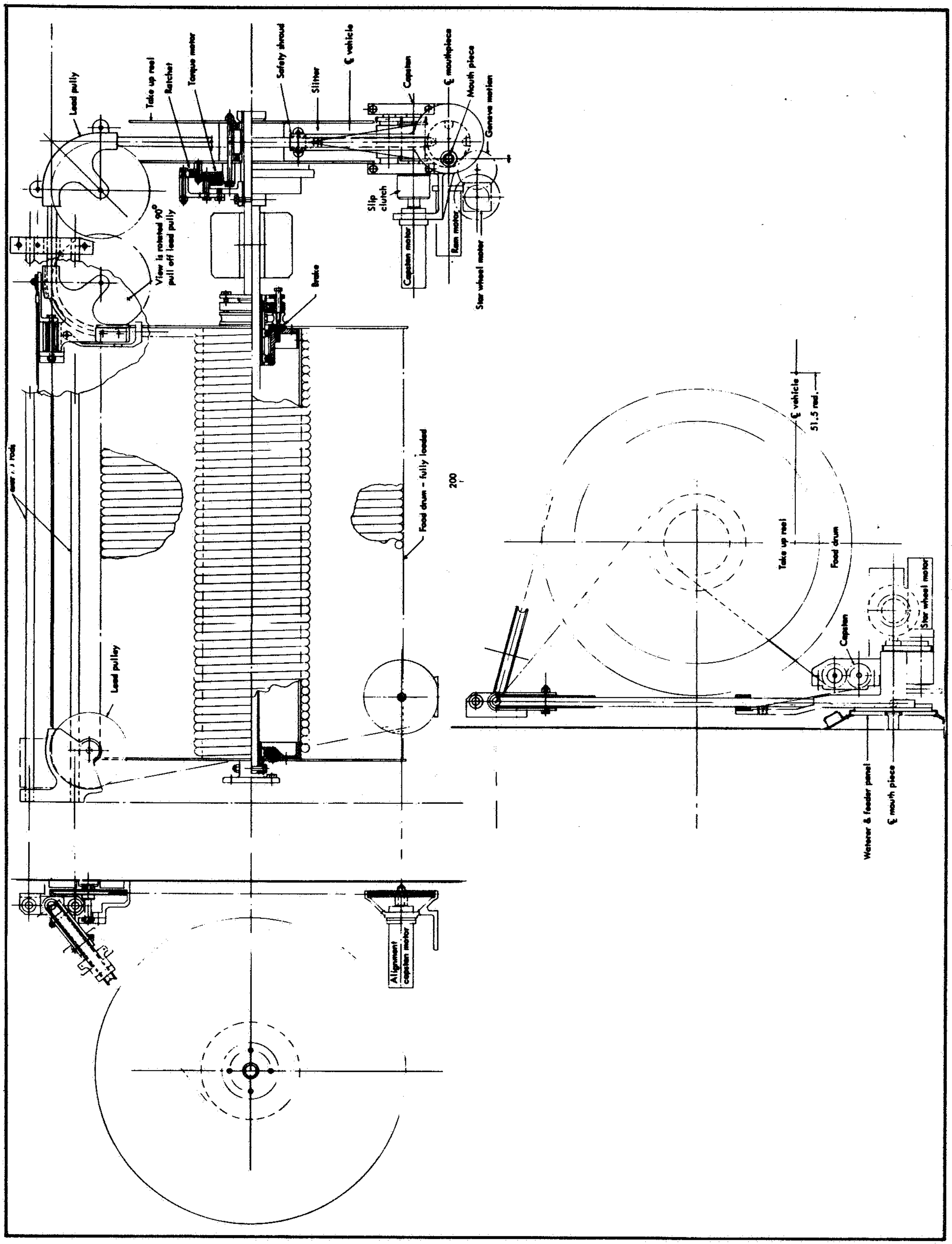


Figure 7. - Feeder

The tube is fed onto a hollow mandril, pushing the pellets into the mandril and down to the delivery mechanism. As the pellets are extracted from the tube and enter the mandril, the tube is opened and flattened into a tape which is then wound upon the storage reel. The prime mover in the system is a calendar type capstan with the two rollers geared together with maximum traction. The rollers are rubber-faced to allow the thicker edges of the tape to pass through while maintaining friction across the full width of the flat tape.

The tube must be advanced one pellet at a time. The geared drive motor will operate the capstan through a slip coupling. As a pellet is shoved into the end of the mandril passage, its presence actuates the capstan switch to the off position. While the motor is being dynamically braked to a stop, the slip coupling will allow over-travel without damaging the pellet or mechanism.

The capstan pulls the tape which forces the pellet into one of four equally spaced notches in the star transfer wheel. As the star wheel rotates, the rim of the wheel takes the residual load of the capstan's indexing tension. A slight ramp on the edge of the wheel makes this feasible without shaving material from the pellet. The principal advantage of the star transfer wheel is to reduce the load and abrasion of the ram used to drive the pellet into the mouthpiece, thereby preventing both urine leakage around the ram and plugging the mouthpiece with feces.

The star wheel is indexed by a Geneva lock wheel driven by a geared motor. The Geneva wheel permits stopping the motor without precise control of motor travel being necessary while accurately positioning the star wheel in readiness for pellet delivery.

Just before the star wheel rotates, the ram is retracted. The star wheel then delivers a pellet in front of the ram. The ram pushes it into the primate's mouth and seals the mouthpiece opening. The mouthpiece presents a smooth, water-tight projection into the living area that cannot be plugged or dismantled, thus reducing the possibility of contamination.

The demand system is comprised of a switch in the mouthpiece, behavioral panel control and related electronics. A clock will actuate the food delivery signals to the food mechanism about one-half second after the animal puts his mouth over the mouthpiece.

Advance development areas: Due to its importance in the success of a mission and its relative uniqueness in terms of prior history, a feeder should be constructed and tested early in the program. Significant factors requiring special attention as part of development will be: the relationship between the slip clutch, capstan motor, and pellet size; adjustment will need to be provided; the "current-on" time of the capstan motor as a function of a given pellet diameter; mandril diameter optimization; and the characteristics of the rolled-up, emptied tube; and the method of using the primate's mouth over the mouthpiece to actuate the feeder operation.

Preliminary equipment list: Major equipment items required to mechanize the preliminary design of the Feeder are listed in table 13. Most of the components as noted are already developed and readily available.

TABLE 13. - FEEDER PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	Planetary gear motor	Globe Industries	102A190-9	4
2	Planetary gear motor	Globe Industries	102A193-11	4
3	Torque motor	Inland Motor Company	T4412	2
4	Zipper tubing	Zipper Tubing Company	Zt-0500	2-2200 ft./roll
5	Cam roll	McGill Company	CF-5/8	2
6	Spur gear	Boston Gear Company	NA 25B	8
7	Spur gear	Boston Gear Company	30B	4
8	External tangent drive	Geneva Motions Corporation	New	2
9	Linear ball bearing	Thompson Industries	A-81420	4
10	Ratchet ring gear	Boston Gear Company	NB-120	2

Waterer. - The concept underlying the waterer design will incorporate the high reliability required for a successful mission. Engineering data shall be gathered for assessment of the operation of the waterer, both for recognition of incipient failure and for evaluation of the system performance during the one year operation.

The waterer should prevent back contamination by desolved material, particulates, or micro-organisms through the delivery nozzle. The water itself will be potable tap water with the mineral content limited to prevent corrosion or scaling. The water will be filtered-sterilized as it is placed in the storage tank. The potable tap water is being used because the primate needs the minerals contained in the water.

The waterer shall provide safe water storage adequate for a one year mission and shall dispense measured amounts of water to the animal upon successful completion of a behavioral task by the animal. The requirements affecting the waterer are:

(1) The waterer shall store 151.5 kg (334 pounds) of water per primate for a one-year mission.

(2) The waterer shall deliver 2 cc aliquots to the animal at the rate of 1 cc/second for an average of 208 aliquots per day for one year without failure.

(3) A switch shall operate when the animal's mouth is on the mouthpiece, the size of which shall be compatible with the animal.

(4) The stored water shall not support or encourage growths of fungi or bacteria.

(5) The stored water shall remain uncontaminated by food, feces or urine present in the Life Cell.

(6) The water shall be delivered to the animal at Life Cell temperatures within ± 5 degrees.

(7) The waterer shall operate in 0 and 1 g field environment.

(8) The waterer design and construction shall be compatible with installation in a spacecraft regarding size, weight, electrical interference, etc.

(9) The waterer shall prevent buildup of contaminating solutes due to corrosion or leaching of component materials.

(10) The waterer shall provide a signal signifying delivery of an aliquot.

(11) The waterer shall tolerate complete sanitization before loading with water.

(12) The waterer shall signal time of delivery to within 0.01 second.

The approach selected for the storage of water is a tank constructed of two aluminum hemispheres bolted together. A stainless steel exit ring enclosed in Viton bladders uses a stainless steel wire mesh to prevent blockage of exit holes. Figure 8 shows the two sets of hemispherical double walled bladders with the exit ring between the bladders. The exit ring supports a 1-square-inch mesh, 0.093 diameter, wire screen. The two sets of bladders collapse simultaneously due to the manifolding of the pressurizing

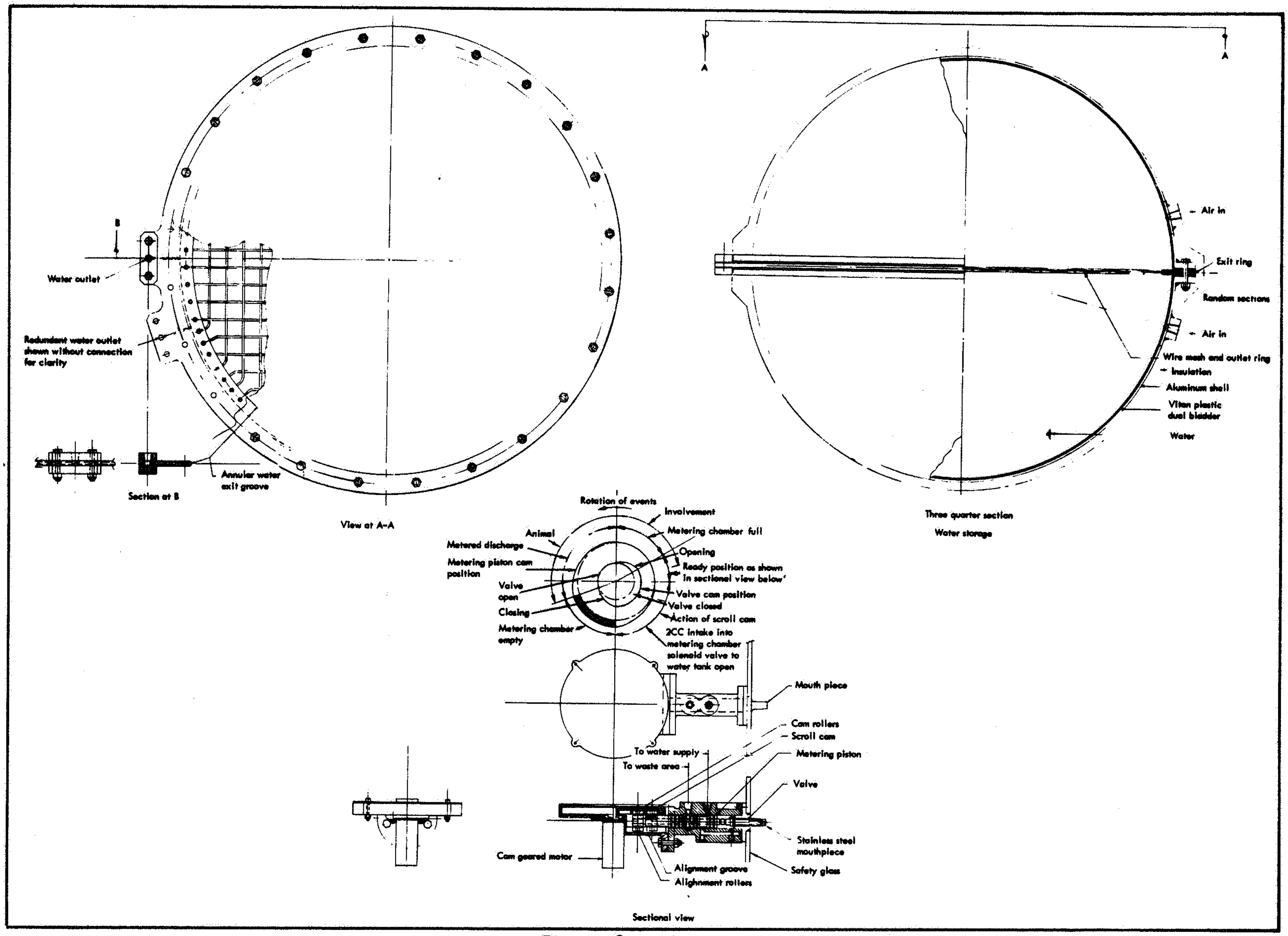


Figure 8. - Waterer

gas, which is introduced into both of the tank hemispheres equally. This ring also provides the option of introducing as many water outlets as is desired to increase reliability through redundancy.

Water from the storage tank must be delivered to the animal in measured amounts at the proper rate. This requires a metering device, valves and mechanism to prevent contamination of the water supply related as shown in figure 9. The method for water delivery incorporates the combinations of valves, metering devices, and a shaft that seals the mouthpiece when it is not in use. Figure 8 shows the details of the delivery mechanism and the selected combination mouthpiece closing device, and metering valve or pump. Both plungers are operated by a combination dual cam driven by a standard geared motor on a predetermined sequence. The solenoid valve is a standard item.

This system using one standard solenoid valve and one standard type geared motor is the simplest and most reliable combination of the approaches deemed feasible. The following sequence represents one cycle of operation: (1) the standard type geared motor drives a cam (machined scroll) which has two milled profiled grooves, (2) the piston idles to the left of the position shown. Upon receipt of a signal the center plug is driven to the left, opening the mouthpiece, (3) then the piston is driven to the right in two seconds, delivering two cc of water to the primate, (4) the motor continues

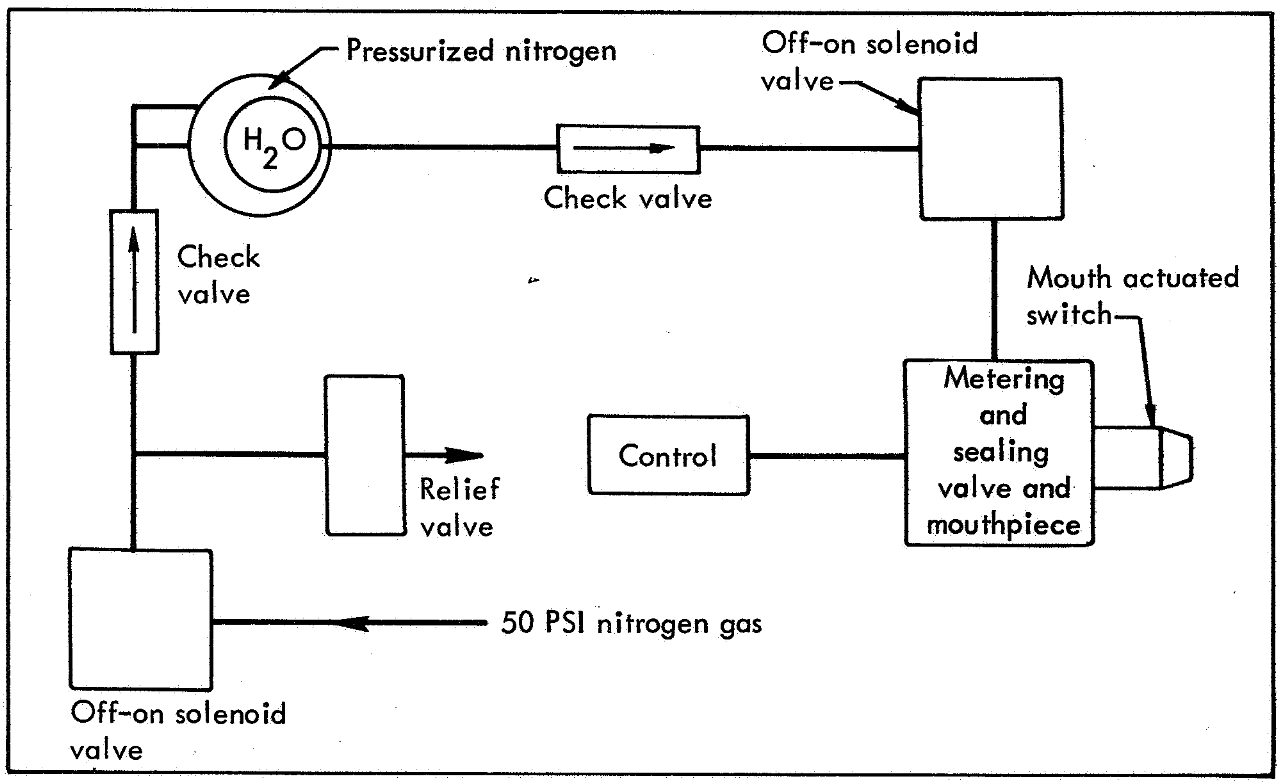


Figure 9. - Primate water supply diagram

to run to seal the mouthpiece opening, (5) the solenoid valve opens, and the motor drives the piston to the left, drawing 2 cc into the cavity, (6) the motor stops, and the solenoid valve closes, (7) the system is then ready for the next delivery cycle.

The shaft and pistons are circularly grooved to reduce leakage and to eliminate side loads which increase operating friction. The small amount of leakage in the valve gland does not warrant incurring the hazards and added complexity of using bellows or fixtures for sealing the movement. A bleed line to the waste management area will dispose of the small leakage that occurs.

Advance development areas: The most common problem involved when moving parts are rubbed together is the use of lubrication. A suitable lubricant may be required to ensure that the moving parts operate during the desired one year mission life. Potential lubricant candidates which could be investigated are as follows:

(1) Determination of a lubricant suitable for one year's service in the spacecraft pressure vessel environment, which would have no deleterious toxic effects on the primate.

(2) Selection of proper materials or combinations of materials to permit corrosion-free storage of water for one year.

(3) Bacterial and fungus growth control in the water delivery mechanism.

(4) Actuation of water delivery by means of the mouthpiece while preserving seal integrity.

Preliminary equipment list: The selection of major component items required in mechanizing the Waterer preliminary design is listed in table 14. As previously noted, the units are already developed and available.

TABLE 14. - WATERER PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	Viton "O" ring	Parker Seal Company	AN6227B	12
2	Planetary gear motor	Globe Industries	102A193-10	2
3	Solenoid valve	Skinner Corporation	V52DA1250	2
4	Ball bearings	New Hampshire Corporation	SR4PP	4

Behavioral panel. - The animals will be trained by the Principal Investigator to perform a number of behavioral tests which are designed to provide measurements of the eating and drinking responses, reaction to frustration and startling, escape/avoidance of noxious stimulation, and circadian patterns of behavior, animal's ability to estimate short time intervals, reaction time, and visual and auditory discrimination, and circadian patterns of behavioral response. A behavioral panel will provide stimulus signals for specific tasks, means for the primate to accomplish the tasks, and means of providing rewards (water and food) for accomplishing the tasks.

The requirements that effect the behavioral panel are:

(1) The behavioral panel shall present stimuli to which the animal must respond in order to obtain food and water.

(2) The behavioral panel shall present stimuli which will cue other specified activity of the animal (e.g., weighting, recovery, etc.).

(3) The panel shall be located away or across from the social window.

(4) The stimulus display and handles should be located for ease of viewing and manipulating when the primate is seated before the panel.

(5) The size, spacing and positioning of stimuli and handles shall be convenient and reduce the possibility of confusion.

(6) Handles projecting 1 inch, 3/8 inch diameter, and moving upward 1/4 inch with a 3-ounce force will be provided for each task.

(7) The food mouthpiece (5/8 inch O.D.) and water mouthpiece (1/4 inch O.D.) shall be located above the handles.

(8) Stimulus lights shall surround the handles and mouthpiece.

(9) The behavioral panel shall be invulnerable to the animal's curiosity and to fouling by waste matter. This includes obstruction of indicators.

Basically, the behavioral panel design shown in figure 10 consists of two panels. The first panel consists of three lights - blue, red, and yellow. The blue light is the stimulus illumination for the timing component (TIM); the red light is the stimulus illumination for the interlock components (ILK); and the yellow light is the stimulus illumination for the vigilance component (VIG). A maximum possible number of eight lights per response panel component are shown. This panel has a translucent material covering the lights and the finger levers whose bases are sealed by an elastomer material.

The waterer-feeder mouthpiece panel constitutes the other portion of the behavioral panel. This subassembly consists of a maximum of twenty-eight white lights, a Waterer mouthpiece, a Feeder mouthpiece, and agian translucent material covering the lights and sealing the interface for the mouthpiece exterior surfaces. Even though the maximum number of lights are shown,

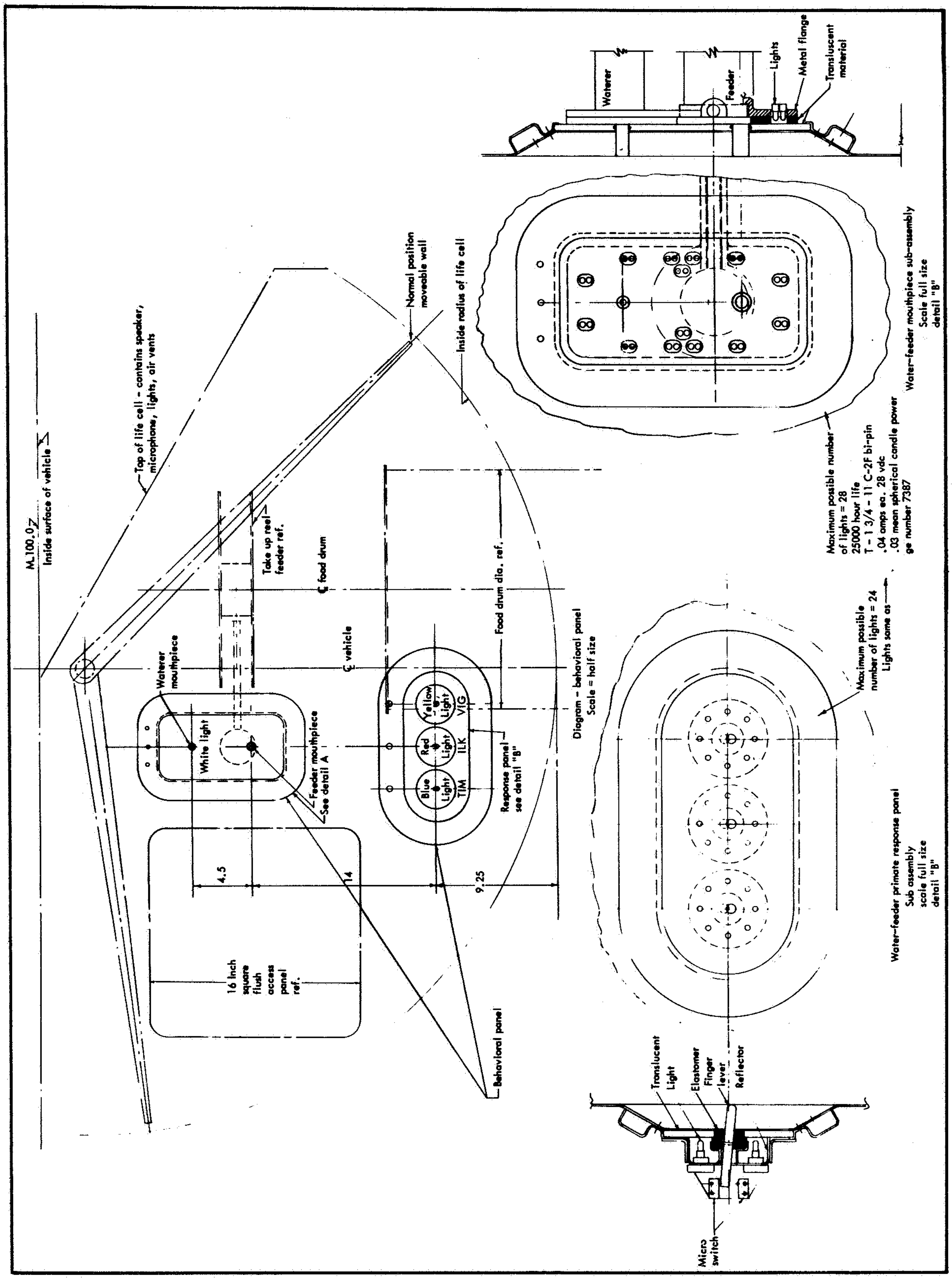


Figure 10. - Behavioral panel

only one-half of the lights will be on. The lights, therefore, have redundancy built into the design.

The behavioral panel will be sealed with Viton gaskets and/or a potting compounds. The translucent material will allow the lights to be seen by the primate but will not allow the primate to touch the light. If polypropylene is found to be scratch-proof, it will be used; otherwise, heat-treated glass will be used.

Advance development areas: Factors which will require additional investigation are summarized as follows:

(1) The number of stimulus lights required to ensure that the primate carries out the required tasks needs to be determined. Inefficient use of these lights increase the spacecraft power requirements.

(2) Materials will need to be investigated to locate a suitable primate-scratch-proof translucent material.

(3) Atmosphere flow distribution will need to be tested to determine an optimum flow pattern to aid in keeping the behavioral panel free of debris.

(4) Elastomer materials will need to be investigated. They provide an important interface between the Waterer or Feeder and the behavioral panel as a sealant.

Preliminary equipment list: Those major items other than the electronics and panel structural material anticipated as being necessary to its mechanization are listed in table 15.

TABLE 15. - BEHAVIORAL PANEL PRELIMINARY EQUIPMENT LIST

Item no.	Description	Suggested manufacturer	Part no.	Quantity per spacecraft
1	Microswitches	Minneapolis-Honeywell	11SM23	12
2	Lamp bulbs	General Electric Corp.	T-1 3/4-11-C-2F-BI-PIN	104
3	Elastomer materials	Kirkhill Rubber Co.	New	6

Environmental control and waste management. - The design analysis effort presented here details the recommended component sizing and arrangement of the environmental control and waste management (ECS/WMS). Synthesis of the spacecraft ECS/WMS is based upon the interrelated aspects of performance, reliability and system interface considerations. Ground support requirements and instrumentation provisions are evaluated following definition of the system configuration and component descriptions. A summary of recommended development areas is included based upon the necessity for qualified hardware operation in 1970.

The ECS/WMS maintains the atmospheric environmental conditions within the primate chamber to assure that the primates will survive for one year in orbit. Beyond this, the control system is required to maintain conditions within somewhat narrower limits than those for mere survival in order that the experimental results may be correctly interpreted. The requirements may be summarized as follows:

(1) Handle thermal loads imposed by the primates and associated electronics.

(2) Control atmosphere level and composition to within specified limits.

(3) Remove primate generated carbon dioxide from the atmosphere.

(4) Control trace contaminants below spacecraft maximum allowable concentration limits established for manned missions.

(5) Maintain a desirable ventilation velocity.

(6) Maintain a desirable effective atmosphere temperature.

(7) Prevent dust from dried feces from spreading through the life support system.

(8) Establish the capability of handling waste material.

The system described below is a baseline system which includes only the hardware components necessary to perform all functions previously selected by trade study analysis. The baseline system does not include redundancy provisions or alternate modes of operation as may be required as a result of reliability evaluations. The baseline system, figure 11, serves as a point of reference in the evolution of a sophisticated system, and will be modified to include recommended provisions of reliability.

Atmosphere circulation: The atmosphere circulation assembly provides primary atmosphere interchange in the life cell; convective cooling of electronic equipment in the pressurized volume; and controlled flow through processing equipment for contaminant control; and thermal and humidity control of the atmosphere. It is composed of 17 functional components, eight of which are derived from existing space qualified hardware.

The synthesis of the atmosphere circulation system involves the diverse functional and performance requirements listed below in table 16.

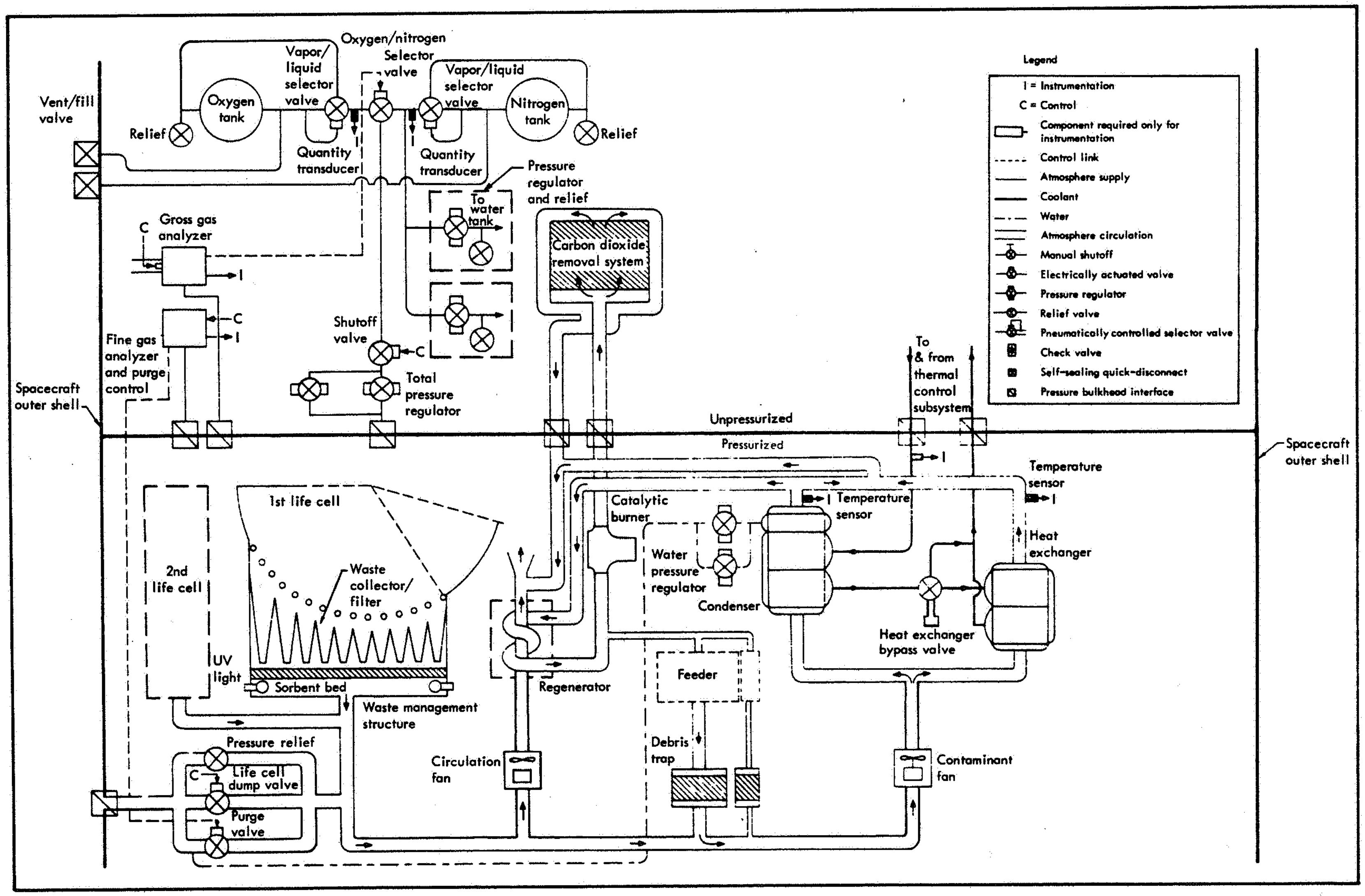


Figure 11. - Environmental control system baseline concepts

TABLE 16. - ATMOSPHERIC CONTROL FUNCTIONS

Function	Required flow rate, lb/hr
Waste collection	2730
Trace gas removal Particulate filtration Bacteria control	All gas circulated through life cell
CO_2 removal	6
Humidity control	82.5
Temperature control	183

The circulation flow values for carbon dioxide and temperature are minimum values necessary to ensure control of the parameter in accordance with the specified tolerance levels.

Of the many concepts for component arrangement, the two-fan approach has been selected over the four-fan approach because of packaging simplicity, weight, and redundancy considerations. The arrangement, shown in figure 12 illustrates the selected flow pattern and generalized component arrangement.

The remaining component arrangement definition involves the carbon dioxide removal equipment. The appropriate trade study has indicated the necessity to provide gas to the LiOH bed in a condition of low humidity and relatively high temperature. The condenser outlet is a source of relatively dry gas, but the low temperature (57°F) is undesirable. A reheater or regenerator, as shown in figure 11, satisfies the conditions.

As shown in figure 11, the circulator fan is the prime motive force of life cell atmosphere interchange. The fan provides atmosphere circulation at a rate of 2458 lb/hr, which, in conjunction with the flow from the contaminant fan, provides a 30-fpm velocity of atmosphere flow through the life cell to facilitate waste removal. The outlet of the circulation fan is directed to the regenerator which is used to preheat the atmosphere flowing to the carbon dioxide removal bed.

Downstream of the regenerator, the flow of the processed atmosphere dumps into the pressurized section of the spacecraft. The circulation of atmosphere in this volume controls the temperature control of all electronic equipment located in the pressurized volume by forced convection.

The atmosphere is drawn to the top of each life cell and enters through flow distribution ports which ensures a uniform velocity and mass flow distribution across the life cell. As the flow proceeds through the life

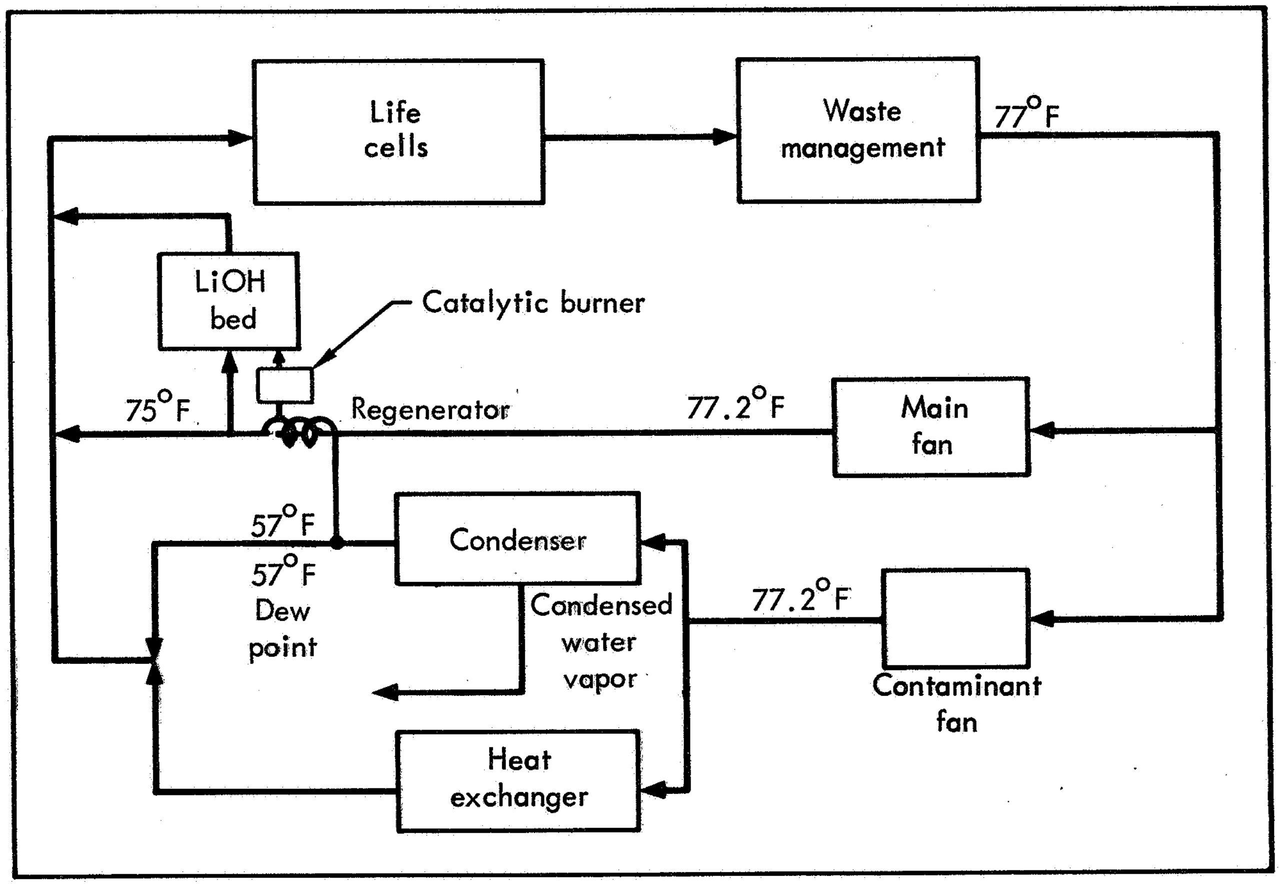


Figure 12. - Baseline atmospheric loop arrangement

cell,debris and wastes are scavenged from the life cell area and transported through the screen bottom to the waste management collector/filter assembly. The velocity of atmosphere flow through the life cell is predicated upon a compromise of fan power and scavenging time. If the debris is to be cleared from the life cell in 5 seconds a velocity of 30 fpm, or a mass flow of 2,730 pounds per hour for the total of two life cells is required.

Wastes, collected by the moving gas stream through the life cell, are transported to the waste management area, and are air dried on the surface of the extended filter. Urine is also retained on the filter, evaporated by the atmosphere flowing through the filter, and removed from the gas stream by the condenser in the humidity control subsystem.

Since the waste collector is a particulate filter, it is reasonable to incorporate all particulate filtration provisions directly into the waste collection system. Further, since the majority of trace gases and bacteria generated in the system will originate in the waste management system, it is reasonable that the control of both bacteria and trace gases be accomplished as close to the source as is practical. The integration of these several functions results in a reduced power requirement for the atmosphere circulation, since the large cross section for atmosphere flow through the filters and beds in the waste management area minimizes pressure drop.

The extended surface of the waste collector/filter is designed to provide 35 square feet of flow area. The wastes collected during a one-year mission require a storage volume of approximately one and one-half to two cubic feet. If wastes accumulate uniformly on the filter, the depth would not exceed 0.75 inches. Because of the large surface area of the collector/filter and probable localization of wastes in a particular area of the collector, the accumulation of wastes over a period of time will not appreciably affect the atmosphere flow through the filter. The relationship of flow to effective flow area has been shown in the trade study on waste management (ref. 6). All urine and fecal water is dried from the collector/filter by the recirculating atmosphere and removed from the atmosphere by subsequent processing.

The absorbent bed, located downstream of the waste collector/filter, removes trace gases, including odors, from the atmosphere. The abosrbent bed is a composite structure containing activated charcoal, phosphoric acid impregnated charcoal, and calcium carbonate,and functions in several ways. The activated charcoal removes most odorous compounds while the acid impregnated charcoal removes ammonia from the gas stream. The calcium carbonate removes most acidic trace gases and the LiOH bed is also useful in removing acidic trace gases.

The absorbent beds are supplemented in controlling trace gases by a catalytic burner located in the flow stream to the LiOH bed. The catalytic burner is particularly adaptable to oxidizing compounds such that the byproducts are removed upon the charcoal or basic beds. Furthermore, the ultraviolet lamp, located in the plenum under the absorbent bed, is one of the disinfectant provisions for the system. The installation of the ultraviolet light in the plenum where gas velocity is low, is particularly advantageous, since it provides the required time to contact with the irradiating field to produce an effective bacteria and virus kill. The only other location within the system where the air is moving slow enough for irradiation is in the life cell itself. However, continued exposure of the primate to the ultraviolet light eliminates this alternative. The lamp, which includes a filter, excludes transmission in the 1843 A band, thereby precluding ozone formation.

The two waste management system plenum outlets form part of the waste management structure, to form a common entrance to the circulation fan where approximately 90 percent of the circulated atmosphere proceeds to the circulation fan. The remaining 10 percent of the atmosphere flow proceeds through a contaminant removal processing loop.

A portion of the atmosphere entering this loop comes from the two food management compartments after passing the two debris traps. The food dispenser may cause dusting of the foot pellets, or even occasional breakage of a pellet. The debris from this source is collected and retained on the debris trap to prevent accumulation in other area of the spacecraft.

A contaminant fan provides a fixed flow of 82.5 pounds an hour through the condenser and 183 pounds an hour through the heat exchanger. Cold coolant, at a controlled, fixed temperature of 55°F and a flow rate of 225 pounds an hour, enters the condenser to cool the gas stream below the dew point and thereby to remove water vapor and control the relative humidity in the life cell atmosphere. The condensed water is separated from the gas stream by a static wick-type water separator which is integral to the condenser. The collected water is vented overboard by the water pressure regulator. The regulator controls the upstream pressure (suction head on water separator) to a value of approximately 1-1/2 p.s.i. below the life cell atmosphere pressure. The suction head on the condenser wicks removes water according to the rate of condensation. The surface tension of the water on the porous wick surfaces balances the reduced pressure applied by the regulator. As water accumulates, surface tension forces become less and the regulator vents water overboard until a balance is achieved.

The coolant leaving the condenser is directed to the heat exchanger bypass valve. Since coolant has been warmed by heat exchange with the gas flow in the condenser, in excess of the dewpoint temperature of the main circulating atmosphere stream, the coolant entering the heat exchanger cannot cause additional condensation. The heat exchanger's only function is to provide variable sensible cooling upon demand. The heat exchanger bypass provides the function of variable heat rejection based upon sensed temperature in the circulating gas stream. The bypass valve is of the wax element, or vernatherm, type. The temperature-sensitive element contains a material which exhibits a high coefficient of thermal expansion. The expansion of the material, in response to temperature change, is used as a direct mechanical force to reposition a modulating flow control valve. The vernatherm-sensing element responds to increasing atmosphere temperature by increasing the coolant flow to the heat exchanger, and thereby increases the heat removal from the life cell atmosphere.

The atmosphere flow from the condenser and heat exchanger join and are vented to the pressurized section of the spacecraft. However, a small portion of the atmosphere flow from the condenser is directed to the regenerator. This relatively dry gas, available at the condenser outlet, is warmed by thermal transfer with the main circulation loop at the regenerator to provide as dry and as warm a gas stream to the entrance of the carbon dioxide removal system as practical. An additional source of thermal energy comes from the catalytic burner located between the regenerator and the LiOH bed. The excess heat from the catalytic burner warms the process gas to slightly above the life cell temperature. The products of oxidation from the catalytic burner are carried with the gas flow to the LiOH bed and retained. The lithium hydroxide bed, which is primarily for carbon dioxide removal, therefore, serves as a postabsorbent bed for the catalytic burner, a main absorbent for acidic trace gases, and as an effective disinfectant.

The carbon dioxide removal system consists of a single bed of lithium hydroxide used primarily as an absorbent for carbon dioxide. The LiOH could form extensive quantities of lithium hydrate ($LiOH\ H_2O$) if relatively cold, wet gas is introduced to the bed which would preclude the removal of carbon dioxide in the atmosphere. The location of the carbon dioxide removal system in the unpressurized volume has been selected for two reasons. First, packaging of the large canister is simplified in the unpressurized section of the spacecraft, and secondly, nearly continuous solar energy is therefore available to warm the carbon dioxide removal system simply by locating the lithium hydroxide canister in the unpressurized section in full view of the sun. The principal disadvantage of the lithium hydroxide absorbent bed to accommodate two primates for a year is a significant weight penalty compared with other carbon dioxide removal systems, such as a molecular sieve. However, the LiOH has been selected because of increased reliability. Figure 13 illustrates the variation in total weight of the carbon dioxide removal system as a function of daily primate carbon dioxide generation rate.

At this juncture, it is well to emphasize the interrelationships between the contaminant control and waste management subsystems with other functions performed within the ECS. The summary of contaminant control provisions for the spacecraft is shown in table 17.

It is also well to remember the unknown factors associated with the exact performance criteria required of a bacteria, or trace gas control system operating with two Macaca primates for one year in a closed ecology. The subsequent development phase, which will involve testing with primates, would yield valuable data necessary to ensure design confidence.

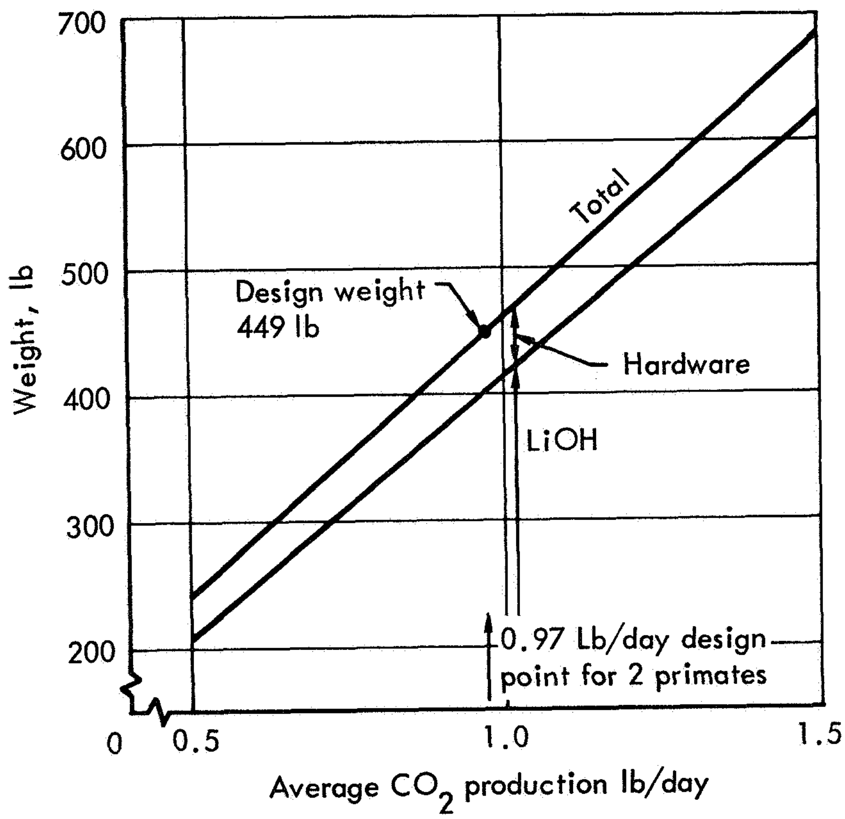


Figure 13. - Lithium hydroxide weight variation

TABLE 17. - SUMMARY OF CONTAMINANT CONTROL

Type	Selected approach
Particles	Mechanical collection
Bacteria	UV Radiation
	Disinfectant (LiOH)
Trace gases	Sorbent beds
	Life cell leakage
	Catalytic burner
Carbon Dioxide	Absorption on LiOH

In order to initiate later development phases with reasonable confidence that the trace gas control provisions are adequate, a catalytic burner is included in the design. The catalytic burner processes atmosphere flowing to the lithium hydroxide bed. The lithium hydroxide bed, therefore, becomes the post-absorbent bed required downstream of the catalytic burner to remove oxidation products formed in the catalytic bed.

Three additional components complete the functional description of the atmosphere circulation subassembly: the pressure relief valve, the purge valve, and the life cell dump valve. The pressure relief valve provides protection of the primate and structure from inadvertent or temporary overpressurization damage.

The purge valve may be opened by command in the event a controlled high leak rate is desired to flush contaminants from the atmosphere. Under normal operating circumstances, this valve would never be actuated. However, in the event of a temporary nondestructive, overheating condition occurring within some electronic equipment, for example, whereupon an upset condition of trace gases could be detected by the gas analyzer, the valve would be commanded to open to supplement trace gas control by the absorbent bed by the flushing action.

The life cell dump valve is actuated only once during the mission. Its use is associated with the recovery operation. After the primate has entered the recovery capsule and the recovery capsule door has closed and sealed, the life cell dump valve is remotely commanded to open. Approximately 15 to 20 minutes is required to vent the atmosphere from the spacecraft. The venting is performed as a safety feature for the EVA astronaut during removal of the recovery capsule, so that the astronaut is not working around a pressurized vessel.

Atmosphere supply: The atmosphere supply assembly stores the major constituents of the atmosphere, oxygen and nitrogen, and regulates this atmosphere within the life cell. Oxygen and nitrogen are added upon demand to maintain the equivalent of a sea level atmosphere within the life cells in terms of pressure and oxygen concentration.

The atmosphere supply assembly is composed of 12 functional components, ten of which are derived directly from existing space vehicle applications. The two new components are the storage tanks for oxygen and nitrogen. Oxygen is stored in the oxygen tank in a low-pressure, 50 psia, subcritical state. The storage function could be accomplished by one or a multiplicity of subcritical cryogenic tanks; however, a single tank is recommended for two reasons. First, a single tank results in a lower penalty than two smaller tanks because of the lower thermal protection penalty per unit weight of stored cryogen. Second, the present spacecraft configuration and packaging constraints permit the use of a single tank for each atmosphere constituent. The selection of a single tank for each constituent of the atmosphere is therefore based upon weight and configuration considerations. The use of two or more tanks and associated isolation valving could result in an increase in mission success probability in the event of a failure of a single-tank system.

The tank contains surface tension devices that orient the liquid at one outlet and the vapor at a second outlet. The vent/fill valve is used prior to launch for filling and topping the cryogen. A relief valve serves as a safety device to prevent overpressurization.

In performing the delivery function, considerations of demand use rate and temperature and pressure conditioning are significant. The equivalent function of a hot coolant flowing through a heat exchanger to warm the cold gas or vaporize the liquid is performed by simply attaching the two cryogenic supply lines to a large structural member. The cryogenic demand use rate results in a thermal load of only 3 Btu/hr to warm the cryogens to approximately ambient temperature before delivery of the gas to the liquid/vapor selector valve. The substitution of thermal inertia for an active heat exchanger offers several advantages. First, coolant leakage at the heat exchanger inlet and outlet couplings is eliminated. Second, the pumping power penalty associated with the coolant flow is eliminated. Third, elimination of even a structural element enhances reliability of the system.

Both the vapor and liquid delivery lines are routed and attached to the large flange of the carbon dioxide removal system to ensure proper temperature conditioning prior to contact with the vapor/liquid selector valve.

Pressure conditioning is accomplished by the vapor/liquid selector valve, as described in the atmosphere supply trade study (ref. 6). The selective withdrawal of either liquid or vapor ensures maintenance of a relatively constant tank supply pressure, even under conditions of widely varying demand rates. The vapor/liquid selector valve serves to selectively withdraw liquid or vapor according to the sensed storage pressure. Vapor is withdrawn when tank pressure increases, while liquid is withdrawn if tank pressure decreases.

Nitrogen is stored in a subcritical low-pressure state, 50 psia, in the nitrogen tank. The nitrogen supply subassembly description is identical to the oxygen supply description above. The oxygen and nitrogen supply subassemblies are joined by the oxygen/nitrogen selector valve. The oxygen/nitrogen selector valve is a two-position, solenoid-actuated valve, which admits either oxygen or nitrogen to the upstream side of the shutoff valve.

The selection of an active system for oxygen concentration control is predicated upon a reliable, long-life oxygen partial pressure measurement device. The development of several gas analysis instruments is currently funded by NASA. Pending successful qualification testing, either a four-channel mass spectrometer or a four-channel ultraviolet-infrared gas analyzer, or both, will be available for this application.

Composition control is accomplished using a mass spectrometer and associated ion vacuum pump. The ion pump is actuated during ground test and all mission phases through launch and orbit injection. After injection into orbit, the ion pump is deactivated and the vacuum of space is utilized by firing a squib which causes penetration of a diaphragm. The mass spectrometer is used because it offers service life in excess of the 1 year which is not possible with a single polarographic sensor, for example.

Four polarographic sensors, having a life expectancy of 90 days could be used, assuming that a sequentially manifolded configuration allowing selective isolation of individual sensors in an inert environment could be provided. In this way, the sensor would be stored inactively in an inert environment until previous sensors had deteriorated and a new sensor was required. A time-oriented command would initiate changeover to the new sensor.

Pressure control is accomplished using a total pressure, aneroid-controlled, demand regulator. The location of this component must be downstream of the oxygen/nitrogen selector valve in order to preclude the necessity of two matched units, one in each of the oxygen and nitrogen supply lines. The oxygen concentration sensed by the gross gas analyzer determines the position of the oxygen/nitrogen selector valve. The total pressure regulator then controls the introduction of the selected constituent to the life cell atmosphere on the basis of total pressure.

A relief valve is included to protect against overpressurization of the spacecraft in the event of a temporary malfunction of the demand pressure regulator. Thus, the purpose of the relief valve is to prevent structural damage to the spacecraft.

The contingency function is satisfied by a purge valve which can be actuated in the event of a temporary trace gas upset condition. The purge valve allows a fixed, relatively high flow rate, exhaust from the life cell atmosphere. The flushing action caused by opening the purge valve stabilizes the atmosphere composition.

A fine gas analyzer capable of detecting and qualitatively analyzing trace contaminants is provided to command the purge valve to open. It is significant to mention that the analyzer's output may be telemetered to a ground station to present an account of the long-term effectiveness of the trace contaminant control provision aboard the spacecraft. If an upset condition occurs and the valve opens to flush the atmosphere of contamination, the telemetered data should indicate the nature and source leading to the upset condition.

The nitrogen supply subassembly provides pressurization of the onboard water supply through the pressure regulator/relief valve. This component maintains a pressure head of approximately 25 psia on the water tank such that water is supplied to the primate waterer upon demand. The valve contains a pressure relief feature to prevent overpressurization.

Nitrogen is used as the pressurant for two reasons. First, nitrogen gas should be less harmful to the bladder within the water tank. Second, the increased demand use rate of nitrogen from the supply assembly tends to make the nitrogen tank less susceptible to thermal variations, which result in pressure excursions within the storage system.

At the conclusion of the one-year mission, when the life cell dump valve is opened to depressurize the spacecraft, the shutoff valve is electrically closed to facilitate the depressurization operation.

ECS/WMS instrumentation: Only three basic types of instrumentation are required for the baseline configuration of the spacecraft environmental control system: quantity of atmospheric gases remaining in the cryogenic tanks, temperature in the atmosphere and coolant loop, and life cell atmospheric pressure. Basically, the instrumentation is used only as a monitor of system performance and, to a lesser degree, to perform minor diagnostic functions. The presence or absence of the instrumentation does not in any way affect the performance of the environmental control system. Therefore, the malfunction of any or all of the instrumentation transducers would not alter basic mission objectives nor would such malfunctions constitute a malfunction or a reliability degradation factor for the spacecraft environmental control system.

Flow rate transducers are located on the oxygen and nitrogen supply lines to provide an indication of the quantity remaining. The data obtained would allow determination of average values of life cell leak rate and metabolic consumption, and could provide some diagnostic information, perhaps corresponding to leaks developing in the spacecraft pressure shell.

Temperature monitoring of the atmosphere and coolant loop allows performance evaluation with minor diagnostic possibilities. The same is true of the life cell pressure data obtained during flight.

The components selected for instrumentation are all derived directly from existing space vehicle applications.

The baseline system as been presented above, consists of all functional requirements to provide environmental control and waste management for two primates in space. This section describes the selected system approach, see figure 14.

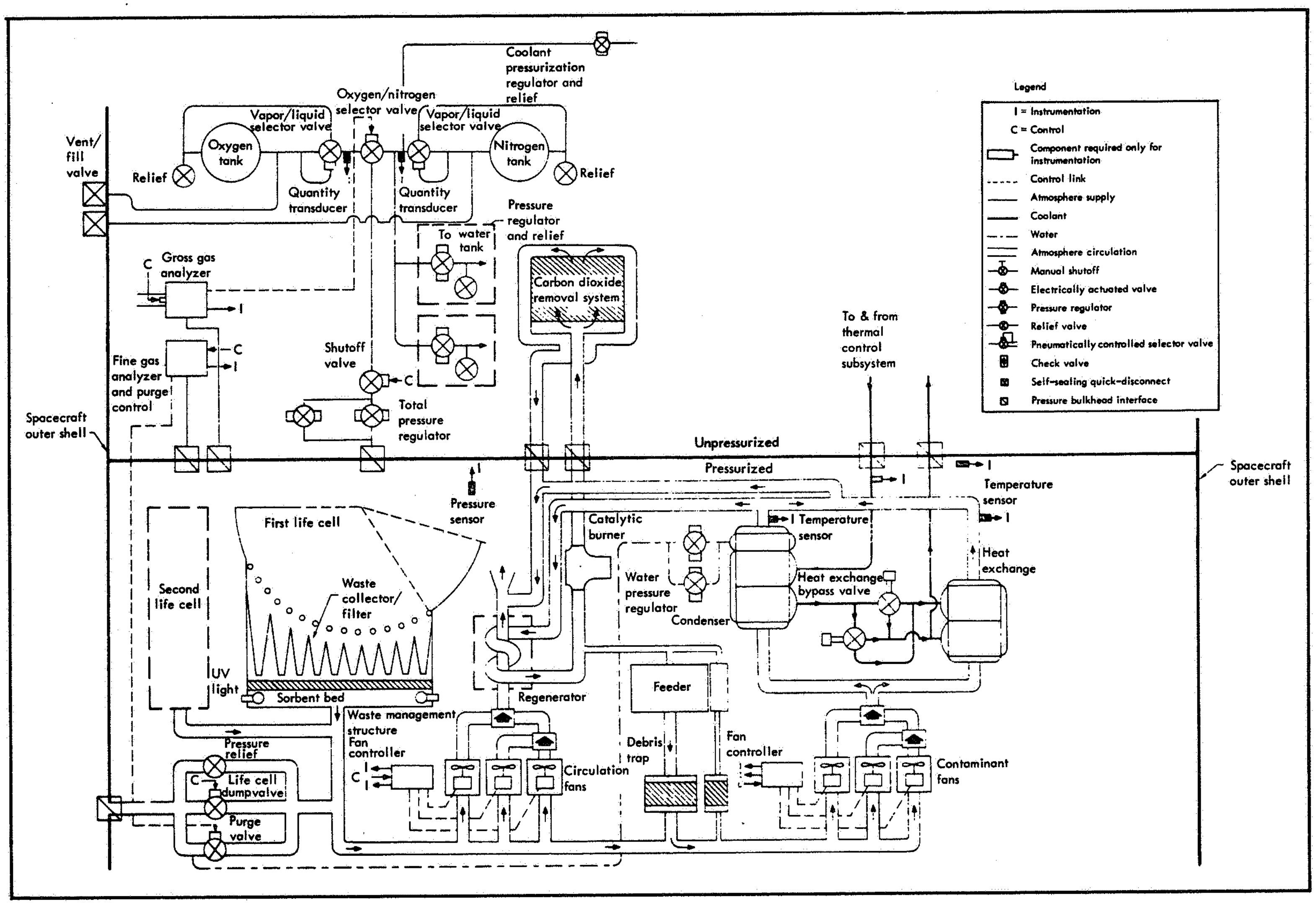


Figure 14. - Spacecraft environmental control system & waste management system block diagram

The differences between the selected system and the baseline concept are due to reliability considerations. Two basic redundancy approaches are illustrated. The most instances the effect is the simple addition of components in an on-line parallel arrangement. The nature of the component allows parallel functions without the necessity to actively switch to a standby component in the event of a failure. Pressure regulators are an example of on-line redundancy.

The fans are illustrations of the second, more complex redundancy approach. The standby units are activated by monitoring devices which interpret the performance of an operative or nonoperating component and provide appropriate control signals in the event of failure.

The detailed description of the redundant circulation fans is identical to that for the contaminant fans, and therefore only one description is presented. The use of redundant fans requires that the performance of the fan in service be verified. This is accomplished by use of a speed monitor, (part of fan controller) which operates directly from the photoelectric commutator of the brushless dc motor. Should the motor slow down, indicating a loss of power or a failure in the motor assembly, the fan controller automatically switches power to a standby fan assembly.

Each of the fan assemblies requires a complement of components performing new functions to satisfy the reliability requirements. For example, in order to switch between fans, a speed monitor is required. In addition, check valves are required to prevent reverse circulation of atmosphere through the respective standby units. The addition of redundant components to the baseline concept therefore results in the assimilation of many additional functions performed, yet at the same time realizing a definite reliability performance increase.

Advance development areas: Two areas associated with trace contaminants require further development effort. The trace contaminant control approach recommended as a result of this phase of study is based upon gross assumptions of a few general trace gases. Experimental investigations of the actual trace gases generated, and their rates, must be conducted using the primate subjects as sources. Of equal significance is evaluation of contaminants generated from the wastes which are collected and stored in direct contact with the atmosphere. Until this data is available, it is difficult to perform meaningful design studies of component sizing for contaminant control. Preliminary weight allocations made during this preliminary design could even be misleading due to the gross assumptions of generation rates.

Another area requiring technical emphasis is in the development of long life, stable, trace gas analyzers. The trace gas analyzer is used both as a monitor and as a control. The control function opens the purge valve in the event of a contaminant upset condition. It is clear that the utility of this function is totally dependent upon reliable operation of the trace gas analyzer. A gas chromotograph, manufactured by MelPar has been recommended for this function. Its life and stability over a one-year period, even with periodic automatic calibration, has yet to be established. It is probable that significant development in this area may be required.

The general approach to waste management includes waste collection by moving atmosphere through the life cell, storage on a extended filter surface, and processing by air drying of the collected wastes. The concept is recommended from several viewpoints; reliability, performance and cost. To verify the integrity of the concept, tests must be performed using the recommended configurations and the primate subjects. Evaluation of the waste management effectiveness is closely allied to the trace contaminant development mentioned earlier, and tests could be performed concurrently.

The control of bacteria within the closed ecology of the life cell merits further investigation. The waste management trade study has discussed the possible application of disinfectants of either liquid or gas to the waste storage area to inhibit bacteria, fungal and viral growths. The possibility of primate sensitivity to disinfectents added to the system was also discussed. This aspect of environmental control should also be investigated by testing with primates and with waste sources as may be experienced in the flight system.

Preliminary equipment list: A preliminary equipment list is included in NSL 67-33, Component Description.

Recovery capsule. - The function of the recovery capsule is to house and support the monkey during the recovery and reentry phase of the mission. The monkey will enter the recovery capsule from the life cell either by conditioned command, or by being forced by the moving wall. The recovery capsule door is closed and sealed and removed by EVA at the end of the mission and stowed inside the CM for return to earth. The sealed capsule must provide ECS support for the animal for up to 24 hours.

The recovery capsule primary requirements are as follows:

(1) House, support, and restrain the monkey during recovery and reentry
(2) Provide atmosphere and thermal control for 24 hours
(3) Ensure safe and expedient EVA
(4) Control insertion of live or decea-ed animal
(5) Preservation of deceased animal
(6) Capable of location inside the Command Module for recovery and reentry
(7) Minimize contamination during one-year life
(8) Minimum entry restriction to the monkey.

Description and performance: The configuration selected for the recovery capsule is shown in figure 15. The significant features of this approach are as follows:

(1) Plug-in self-contained active environmental control system for each capsule

(2) Interior volume of 1.5 cubic feet with padded walls

(3) A port hole window for astronaut observation of specimen

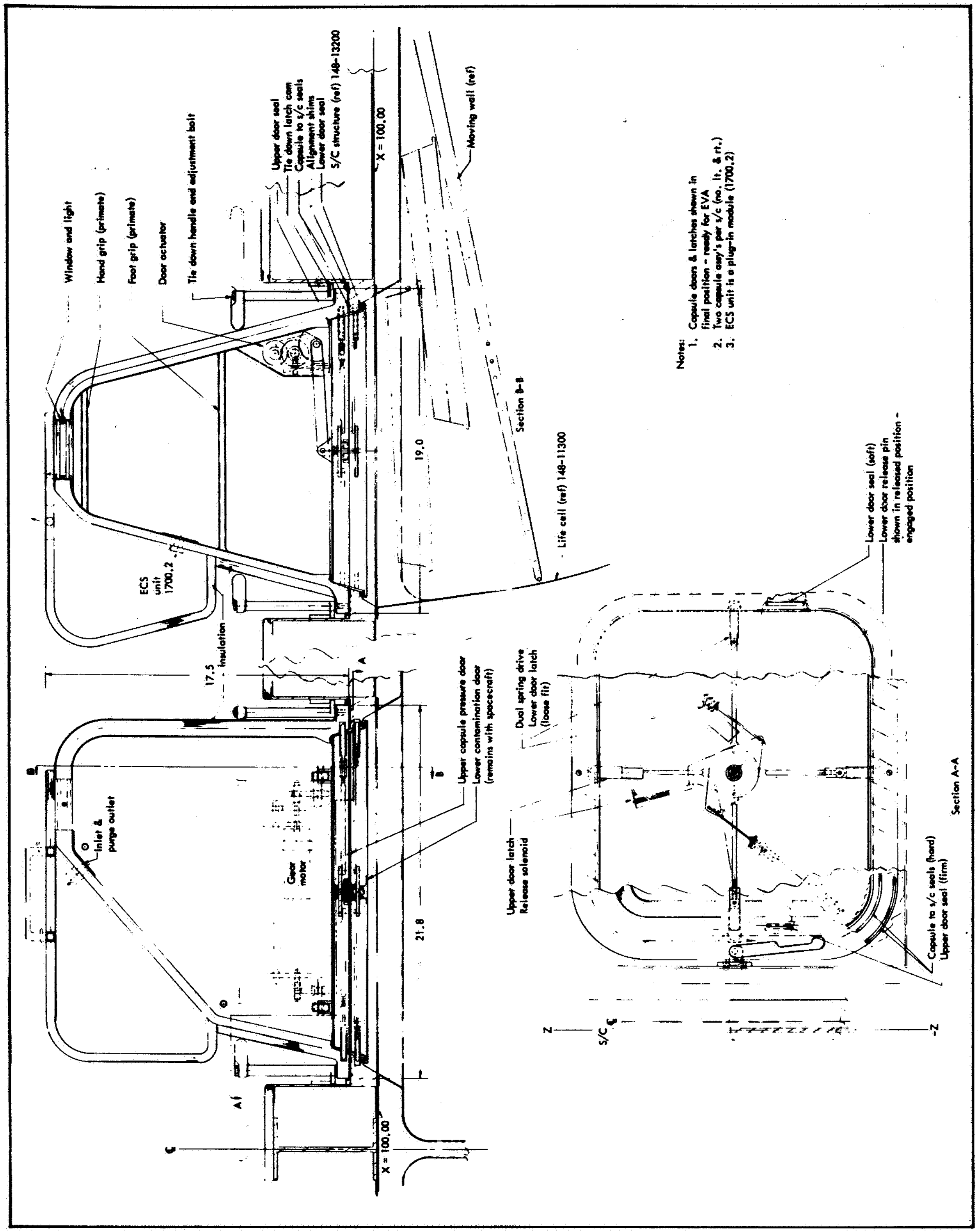


Figure 15. - Recovery capsule

(4) Double sealed vault type door closure to assist in insertion of animal into capsule; to seal cage opening when the capsule is removed; and to eliminate the need for a contamination cover

(5) Quick release tie-down latches - also used for CM mount

(6) Double capsule to spacecraft seals

(7) Expedient and safe EVA operation

(8) Preservation and recovery of dead animals.

The external overall dimensions of the capsule are approximately 20 inches wide, 22 inches long, and 17 inches high. The exact external shape of the capsule can be modified to suit the CM storage requirements once these criteria have been defined. The sides of the pressure wall are sloped to provide a tapered inner volume for animal orientation and positioning. The sloping sides shown in figure 15 conform with the CM floor and wall geometry behind the crew couches. The recovery capsule environmental control subsystem is shown as a separate module which plugs into this pressurized structure.

The plug-in recovery capsule environmental control subsystem is used during the 24-hour recovery period, starting with the primates entering the recovery capsule, until the primate is released from the capsule into an isolation environment on earth. Figure 16 is a schematic of the recovery capsule environmental control system. The recovery capsule environmental control subsystem provides appropriate controls for the environmental parameters of humidity, temperature, carbon dioxide, trace gases, oxygen composition, total pressure, and particulate filtration.

The recovery capsule ECS operates in three basic modes. It operates while attached to the spacecraft, where spacecraft commands and electrical services are available. The second mode is operation during transit between the spacecraft and Command Module, where electrical service is supplied by a tether, umbilical or a plug-in battery unit brought from the command module. And the third mode is operation aboard the command module.

The first mode of operation is initiated remotely from a ground command. The Recovery Capsule ECS oxygen shutoff valve is opened and the circulation fan is activated. Opening the shutoff valve allows the total pressure regulator to provide make up gas to the environment as required by demand. Atmosphere storage is accomplished by the use of a high-pressure gas storage bottle.

The fan circulates the atmosphere through the evaporator and the heat exchanger, neither of which provides a useful function during initial phases of the recovery operation. The circulating atmosphere enters the Recovery Capsule, mixes with the atmosphere, and returns to the RC/ECS through the second disconnect. From the disconnect the flow enters the contaminant removal filter, which performs the following functions: a mechanical

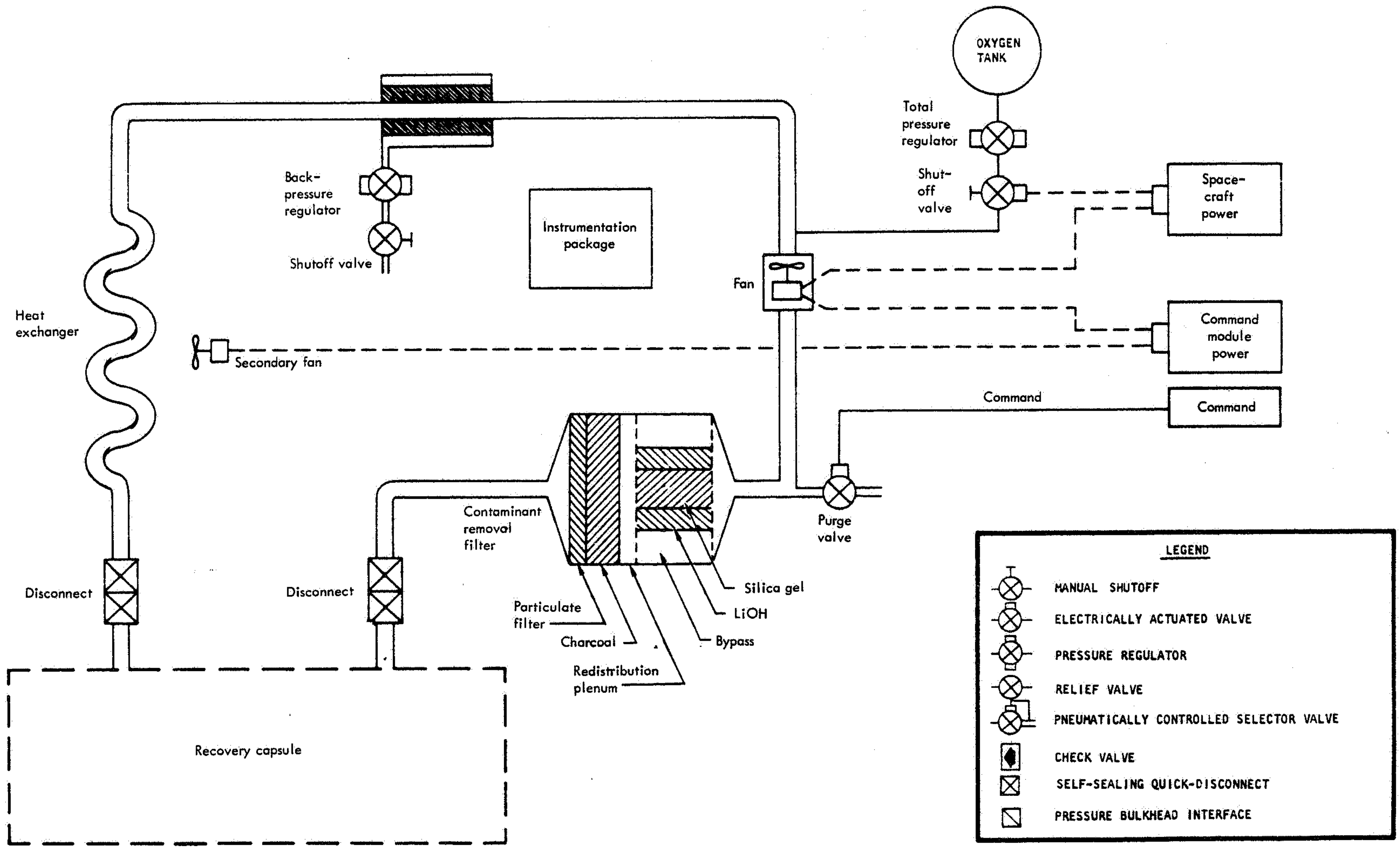


Figure 16. - Recovery capsule ECS block diagram

filter provides particulate filtration of the atmosphere as the first process of purification of the total gas flow; filtration is followed by full atmosphere flow through an activated and impregnated charcoal bed to remove odors and most trace gases; partial flow of the circulating gas is directed to silica gel bed, for humidity control, arranged in parallel with a lithium hydroxide bed, for carbon dioxide control.

The description presented up to this point represents the recovery capsule ECS operation while the recovery capsule is still located aboard the spacecraft, and prior to astronaut EVA. When the astronaut, performing EVA, readies the recovery capsule for transfer to the Apollo Command Module, the astronaut must perform the following activities to ready the ECS for the second mode of operation.

The electrical service to the Recovery Capsule ECS must be transferred from the spacecraft to the electrical umbilical/tether originating in the CM. Concurrently, the astronaut manually overrides the electrical solenoid activated oxygen shutoff valve such that the shutoff remains in the open position even without electrical power applied. The same override operation described can be used to open the evaporator shutoff valve, thus enabling the water evaporator to provide thermal control while the recovery capsule is in transit to the CM. Thermal control prior to this phase of recovery was accomplished by conduction to the spacecraft. While the recovery capsule is in transit, the circulating gas flowing through the evaporator is cooled by convective heat transfer to the low-pressure water evaporation system. The temperature of evaporation is controlled by the backpressure regulator. The regulator maintains a fixed evaporator pressure of 0.35 psia, corresponding to a 70°F saturation temperature, and in so doing controls not only the temperature to allowable limits but can accommodate fairly large variations in heat rejection requirements. During the transit phase, the heat exchanger and secondary fan are still inoperative.

The third and final mode of operation begins as the recovery capsule is stowed aboard the Command Module. As the Command Module is repressurized to 5 psia, the water evaporator no longer provides thermal control because of the increase in evaporator backpressure. The heat exchanger and the secondary fan are used to provide thermal control during this final recovery phase. The atmosphere within the command module is circulated through the heat exchanger to affect temperature control during this final recovery phase. This method of cooling is used until the primate is removed from the recovery capsule.

The recovery capsule includes a purge valve used only in the event that a deceased primate is to be isolated from the second primate and the recovery capsule atmosphere vented to space. The purge valve would remain open until the arrival of the EVA astronaut. Upon disconnect of the electrical service from the spacecraft, the solenoid valve would close and remain closed during the remainder of the recovery operation. Closure is desirable to preclude the possibility of astronaut and/or CM contamination.

The instrumentation package shown in figure 16 monitors the following parameters:

(1) Oxygen pressure
(2) Water quantity
(3) Fan speed
(4) Secondary fan speed.

The inclusion of this element is not to satisfy reliability per se, but rather to verify operational readiness of the ECS prior to the time of recovery. The Recovery Capsule ECS is filled with high-pressure oxygen and water at time of launch. Both expendables must be retained during a one-year storage period prior to the 24 hour operational recovery period. Leakage is, of course, possible during this time.

The parameters of pressure and quantity are monitored during the one year storage period to evaluate the condition of readiness at time of recovery. When the Recovery Capsule ECS is initiated, the fan speed indicators provide the final assessment of reliable operating conditions.

The reliability analysis indicates that even with nonoperating units, safe recovery of the primates can be achieved by use of a single ECS shared by the two recovery capsules, or by a spare replacement brought along in the CM at the time of recovery. Without a monitoring function aboard the ECS, the exact status of the unit is not known until the recovery operation has been initiated. The recovery operation could be greatly simplified if the status of the ECS units is known several weeks in advance, and EVA operating procedures modified accordingly.

The individual, self-contained ECS concept allows the ECS to be removed as an entity from the Recovery Capsule, and replaced if necessary. The replacement could even be accomplished at the time of recovery by the EVA astronaut. If replacement is not possible, both recovery capsules may be connected to a single ECS as stated above. The degradation of performance, when operating at twice load, is only in temperature and humidity control while in the CM. Oxygen and carbon dioxide control will be accomplished without degradation during this contingency mode. The selected approach provides the greatest flexibility, primate safety, and reliability of recovery.

The interior of the capsule shown in figure 15 is designed to orient the animal in the desired supine position for reentry. This same supine attitude also positions his feet and arms away from the capsule door to ease the closing operation. This orientation is accomplished by placing a porthole in the far corner of the capsule and by sloping the walls, so that the primate cannot look through the port hole unless he is in the proper position. In addition, two handlebars are located, one for his hands and one for his feet, to further position him in a natural position of facing upward. The window may be exposed to space at the time he enters the capsule to give him an outside view, or it may be found by test that it should be covered by insulation prior to the EVA recovery task. The insulation cover in such a case could be painted with some attractive design and illuminated by a small bulb placed between the glass panes to attract the primate into the desired position.

The primary purpose of the port hole is to enable the astronaut to observe and attract, if desired, the primate to the proper position for closing the capsule door and for observation during the reentry mode. While it is felt that this window should be enticing to the primate, it can be deleted if the opposite effect is noted during tests. It could also be provided with a cover for removal by the astronaut at his discretion.

An additional feature of the window is related to inflation of cushions. Should tests indicate that an inflatable cushion or back rest is desired, it could be easily incorporated and stored prior to use in the wall of the capsule. When the astronaut looks through the window and sees that the animal is in the proper position, he would inflate the cushion behind the monkey. In this way the possibility of smothering the animal due to improper position is eliminated. The cushion will not interfere with the air ports because the fresh air inlet for the capsule is located behind a large mesh panel near the port hole, and the air outlet is located behind a large mesh panel behind the handlebar grip for his feet.

To satisfy the above requirements the inside of the capsule and door will remain sealed and unexposed to the animal or cage prior to the recovery operation, since contamination or damage of these critical seals would most likely occur upon repeated exposure. The vault type door selected provides a continuous seal and animal insertion assistance with a simple actuation mechanism and a hinge which is not exposed to the cage. It also integrates well with the moving wall device to forcibly move a living or deceased animal into the capsule. Due to the smooth exterior of this door, a second exterior door can be easily attached to the primary door. This second door serves as a seal between the capsule door and the cage during the one year mission and thus keeps the primary door surface and its "O" ring seals free of contamination or animal damage. When the capsule is removed during EVA, this second door is automatically released from the primate door and remains latched in place, thus closing off the cage opening. This prevents debris from floating out of the cage and also eliminates the need for adding a contamination cover to the capsule.

The recovery capsule is removed by the astronaut during EVA after being in space for one year. It is attached and sealed to the primate spacecraft to prevent the loss of atmosphere across a 14.7 psi pressure differential. Thus the seals and attachment mechanism are required to be easy to release and extremely reliable. Fastening devices selected are cam type hatch latches. The hatch latches are very simple devices mounted on the capsule structure. To release them, the astronaut turns each of four handles 90 degrees by exerting a reasonable torque, which can be set at the time of installation. With this design it would be impossible for the astronaut to generate enough torque to release the handles, if the spacecraft internal pressure had not been vented to space. This is a safety feature as it would be very dangerous to release the capsule while the spacecraft was pressurized, since a force of approximately 5,000 pounds would separate the capsule from the spacecraft.

These same latches serve to attach the capsules to their storage mounting plates located in the CM. The CM mounting plate also serves as an additional pressure cover to prevent any capsule leakage. This mounting plate/cover would have electrical interface connectors attached thereto to automatically provide interface hookups. Rapid capsule removal from the CM could be accomplished by releasing the EVA latches. Normal removal would be accomplished by unbolting the mounting plates from the CM to retain the added pressure plate seal, until the capsule was delivered to the laboratory for animal removal.

Should a specimen die during the mission, it is desirable that he be moved into the recovery capsule. After sealing off the capsule, it would be vented to space, thereby vacuum preserving the specimen for examination upon return to earth.

If the animal's terminal illness were detected in time, a decision could be made to move him into the recovery capsule prior to death. This would ease the capsule entry problem if he would cooperate during this task. After sealing the capsule door, the venting operation would be performed. Should he be incapable of assisting in his transfer to the capsule, the insertion would have to be forced. In order to force the animal into the capsule, a moving wall or cage-sweeping device is required. This device is shown in figure 17 and would move the specimen to the opening of the capsule at the end of the cage. In this position the previously opened capsule door swings along the moving wall's surface to get behind the specimen and push him into the capsule. Should the specimen's tail, foot, or hand get caught in the door, the following procedures could be employed:

(1) Recycle door action, partial or full stroke. This should accomplish the job since specimen is just floating and the odds favor prompt closure.

(2) Back door off slightly and put spacecraft in a slow rotation, which will move specimen to far end of capsule and allow door to close.

(3) Back door off slightly and open the capsule vent to space. The rush of air through capsule door should draw in the extremity and move the primate to the vent screen at the far end of the capsule.

Note: This last method will reduce the atmospheric pressure of the entire spacecraft and could affect the other primate, depending on the drop in pressure. The ECS atmospheric supply will make up the lost air.

Advance development areas: The major question remaining on the recovery capsule is the size, shape, weight allowance and location available inside the CM for the reentry mode (ref. 6). Naturally, this requirement could pose a serious constraint on the size and configuration of the recovery capsule, its environmental control system, and could in fact affect the control spacecraft itself. The resolution of this unknown should thus receive the highest priority in the next phase of the program.

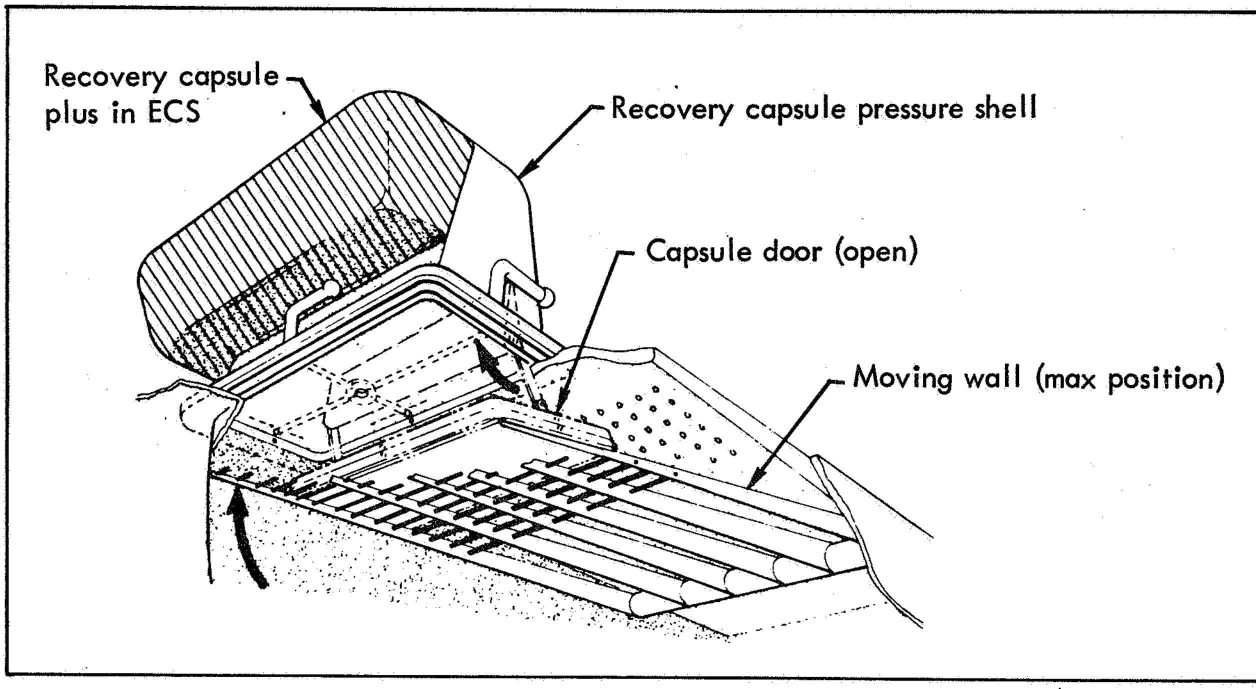


Figure 17. - Recovery capsule door closure approach

Preliminary equipment list: Table 18 lists the major equipment items, other than the environmental control system which is listed in Component Descriptions, (ref. 9).

TABLE 18. - RECOVERY CAPSULE PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	Pressure envelope	Northrop Corp.	New	2
2	Dual door assembly	Northrop Corp.	New	2
3	Door actuator assembly	Northrop Corp.	New	2
4	Gear motor	Globe Industries	102A152-11	2
5	Door latch assembly	Northrop Corp.	New	2
6	Cam latches	Northrop Corp.	New	8

Thermal Control

The purpose of the Thermal Control Subsystem is to establish a balance between the internal heat generation in the spacecraft and the external solar, planetary, and albedo radiation, such that the equipment and primates remain at temperature conducive to successful execution of the spacecraft mission. Figure 18 schematically illustrates the function of this subsystem. Mission success and primate safety greatly depend upon the Thermal Control Subsystem. Therefore, the design of the Thermal Control Subsystem reflects the reliability required for a successful mission.

Thermal control requirements. - The Thermal Control Subsystem provides heat rejection and control for the environmental control subsystem of the Life Support System as well as the electronic subsystems incorporated within the spacecraft. Requirements for this subsystem may be summarized as follows:

(1) Shall have a life of one year

(2) Shall be capable of handling heat loads and variations while maintaining temperatures within current design limitations

(3) Shall control the life cell effective temperature within limits of 73 ± 4°F (dry bulb temperature tolerance is even smaller) for an air velocity of ten feet per minute

(4) Maintain temperatures of the spacecraft's system within limits during checkout, prelaunch and launch phases.

Thermal control description and performance. - Due to the different temperature requirements imposed by the Life Support System and the electronics located outside the pressurized portion of the spacecraft, the Thermal Control Subsystem is divided into two subsystems - the active thermal control subsystem and the passive thermal control subsystem.

Active thermal control subsystems: Due to the strict temperature limits imposed by the Life Support System, an active thermal control subsystem has been selected to handle the heat rejection requirements from the Life Support System. The system described below is a baseline system which includes only hardware components necessary to perform all functions previously selected by trade studies analyses. This system shown in figure 19 does not include redundant provisions or alternate modes of operation that may be required as a result of reliability evaluations.

Thermal control is accomplished using a liquid coolant loop. The selection is based upon the following criteria: low power, sensitive control response, and flexibility to variations in heat rejection requirements. The coolant flow rate selection is shown in figure 20 with the condenser and heat exchanger arranged in series. The design flow is established at 225 pound per hour, based upon FC-75 as the coolant. The dashed line in the figure corresponds to minimum thermal rejection and falls below the minimum condenser outlet tempera-

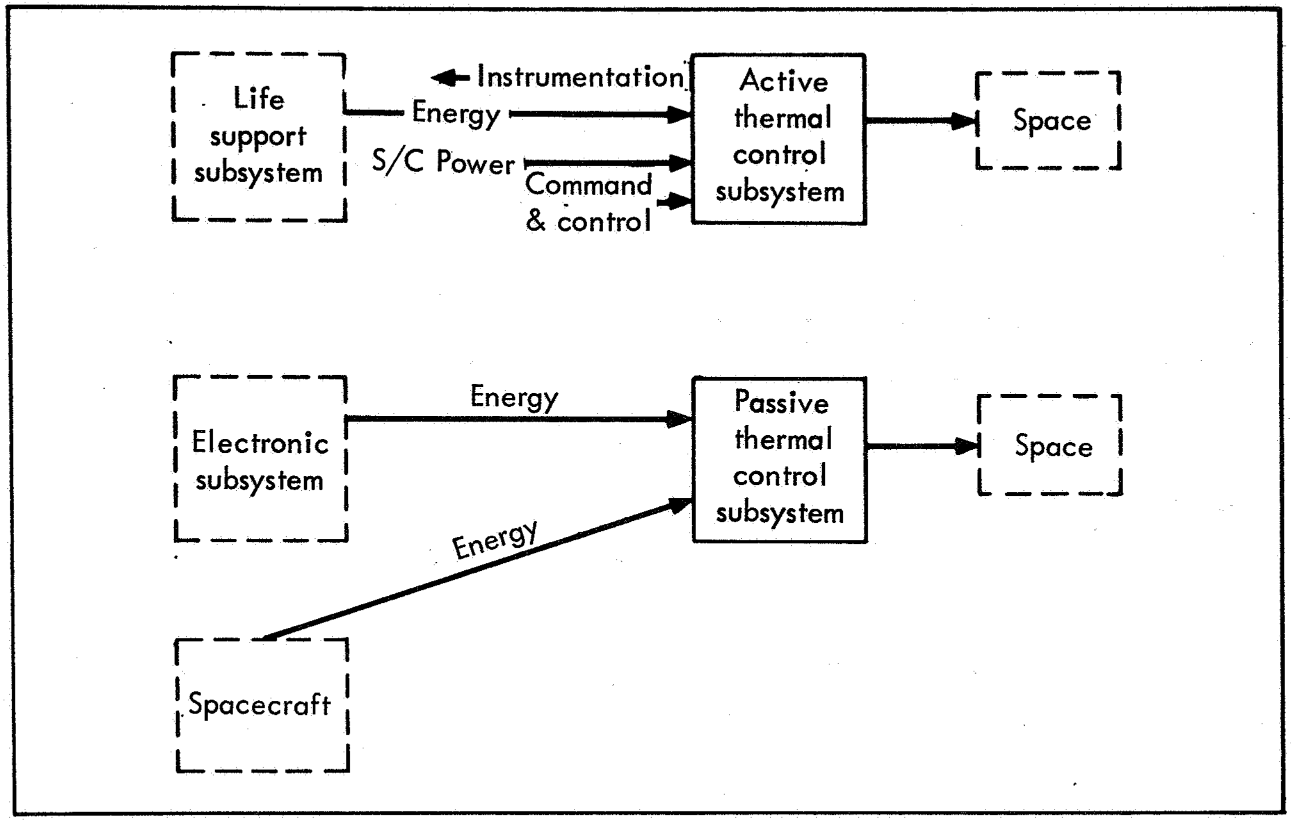


Figure 18. - Thermal control subsystem block diagram

ture. However, this condition is satisfactory, since the latent load is low (resulting in a lower dewpoint) and the coolant flow is directed to bypass the heat exchanger, since the condenser provides all required sensible cooling. Under conditions of higher loads, the coolant temperature of the condenser outlet increases with increasing life cell dewpoint, and therefore precludes the possibility of condensation in the heat exchanger.

Significant criteria for selection of a particular coolant are low pumping power and low freezing point. After evaluation and detailed analysis of the radiator itself, FC-75 was selected as the coolant for this application. The selection was based in part upon radiator temperature control using regenerative techniques. The description of the two most applicable approaches, bypass and regenerative, was presented in the Thermal Control Trade Study and the ECS Thermal Management Trade Study (ref. 6).

Included in the thermal control coolant loop assembly are eight different components, five of which are directly derived from existing spacecraft hardware items.

The pump assembly provides the flow and pressure head requirements for FC-75 circulation. The pump assembly consists of a centrifugal pump and an accumulator. The accumulator provides a small reserve of coolant to compensate for leakage and for thermal expansion. The accumulator includes a spring and bellows combination for pressurization of the coolant loop and to provide a net positive suction head at the pump inlet.

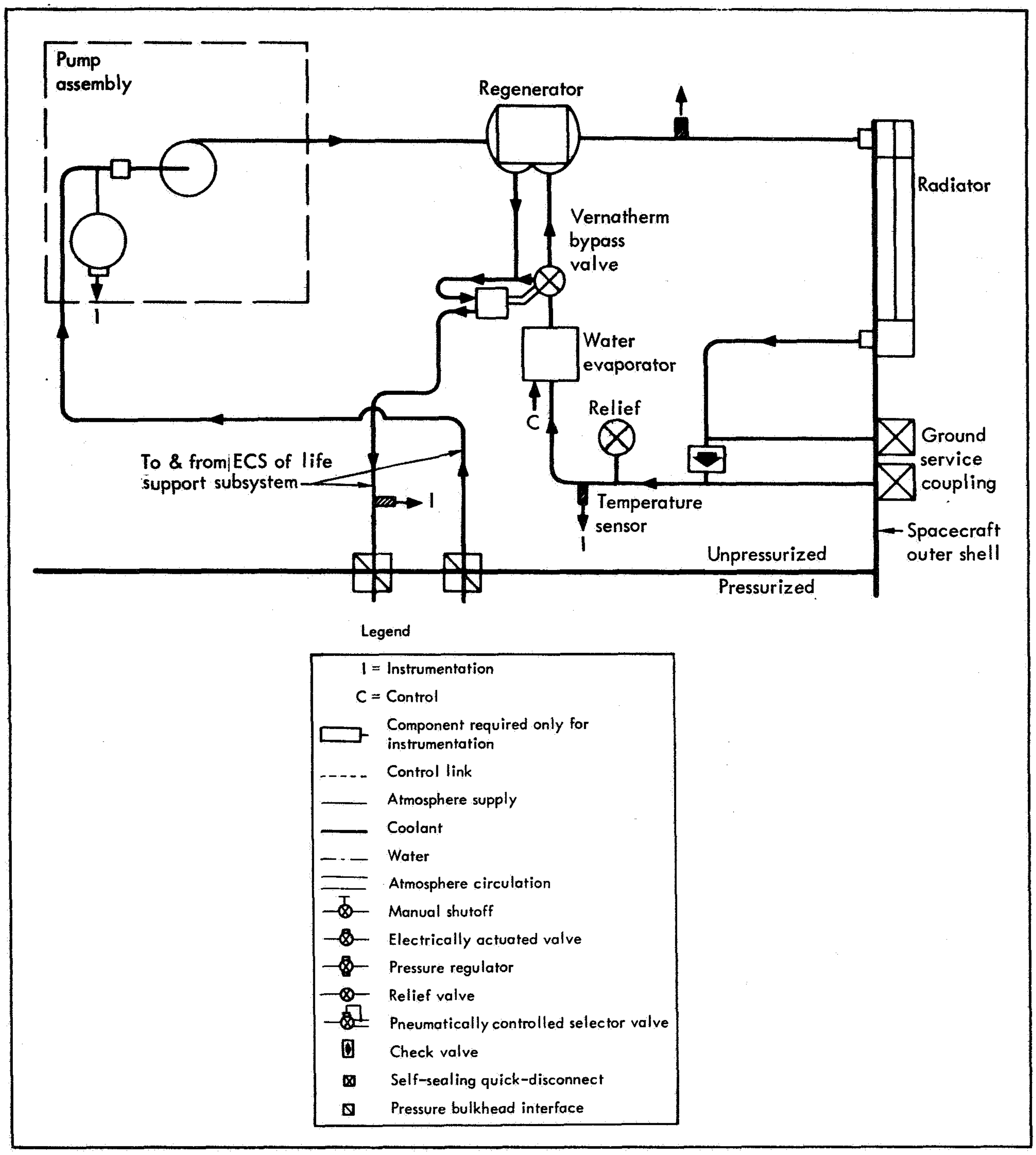


Figure 19. - Baseline active thermal control subsystem

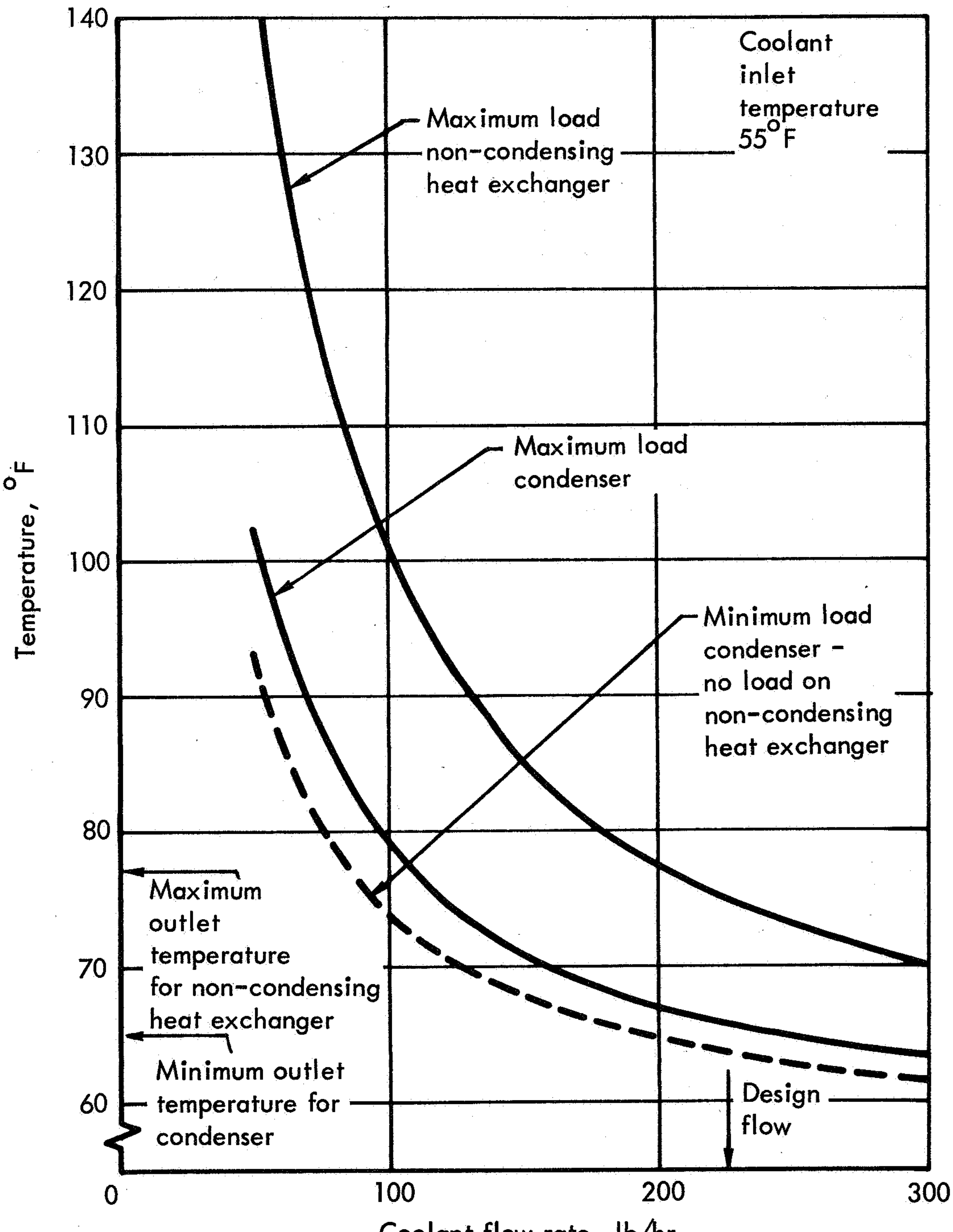


Figure 20. - Thermal control subsystem performance

From the pump assembly, the coolant flows to the regenerator where the full coolant flow passes through the regenerator and is directed to the space radiator. The radiator consists of 20 square feet of surface with two tubes running along the circumference of the spacecraft exterior. A fin efficiency of 90% was imposed on the radiator design. A relief valve located at the radiator coolant outlet, serves as protection against structural damage due to overpressurization. The check valve operates in conjunction with the two ground service couplings. During ground testing, the radiator will be ineffective as a heat sink and a ground cooling service cart will be necessary to provide the heat sink function. Coolant from the service cart will enter the coolant loop through the service coupling downstream of the check valve. The check valve will be closed because of the higher pressure at the entrance of the coolant loop. Coolant will be circulated by a pump in the service cart. After passing through the spacecraft loop, the coolant will return to the service cart through the service coupling located upstream of the check valve. The warm FC-75 will then be cooled prior to return to the spacecraft.

A water evaporator is included for use during the 3 hour period after liftoff and injection into orbit when the space radiator will be nonoperative due to the thermal shrouding effect of the SLA and/or ATM Rack. The water evaporator is a self-contained unit composed of a wick-type evaporator, back-pressure regulator, water reservoir, and start valve. The reservoir contains three pounds of water, sufficient for a minimum of three hours of operation.

A vernatherm valve is used to control the coolant temperature flowing to the condenser. The valve operates in the following manner. The wax-sensing element is in intimate contact with the coolant flowing to the condenser. If the sensed temperature is less than the 55°F desired, a portion of the coolant from the radiator is warmed by flowing through the regenerator. The warmed coolant is then mixed with the coolant from the radiator outlet, thus achieving a mixed temperature closer to 55°F. On the other hand, if the sensed temperature prior to the condenser becomes higher than 55°F, less warm coolant from the regenerator is mixed with the radiator return, thus lowering the condenser inlet temperature.

The design flow of 225 pounds/hour of FC-75 coolant satisfies the conditions of heat rejection shown in table 19 and noted previously on figure 20.

Figure 21 presents the baseline system as augmented for reliability purposes. Two basic redundancy approaches are illustrated. The pump assemblies are illustrations of the first approach. The standby units are activated by monitoring devices which interpret the performance of an operative or non-operating component and provide appropriate control signals in the event of failure.

The second redundancy approach involves the coolant loop and specifically the radiator. It is quite unlikely that the radiator would fail except due to a penetration of a meteoroid. Such a penetration would result in coolant leakage and eventually loss of thermal control and the life support system. A monitoring system is provided which insures that any leakage is detected, a new radiator coolant path is enabled, and the coolant loop refilled with coolant.

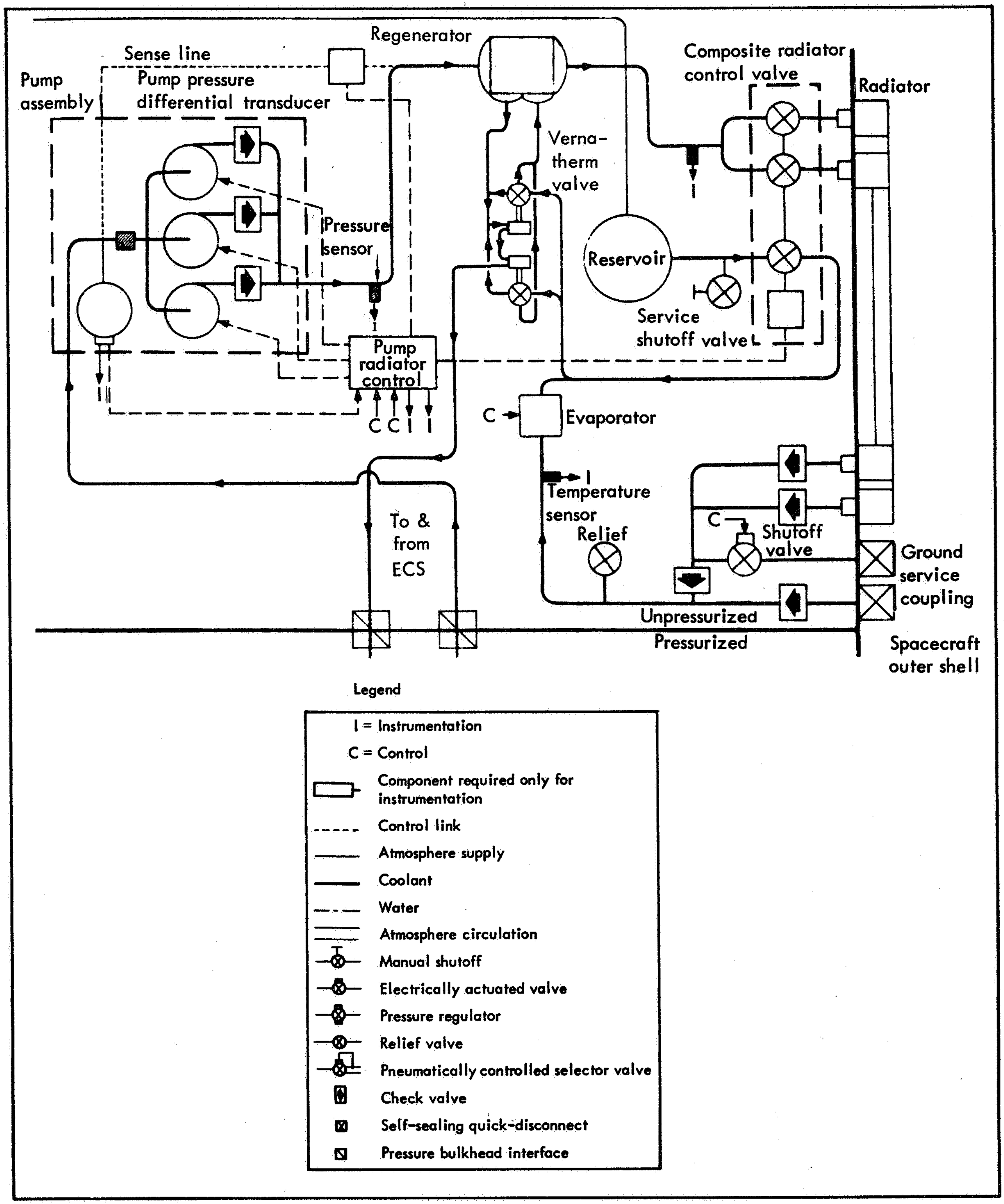


Figure 21. - Selected active thermal control subsystem

TABLE 19. - LIFE SUPPORT SYSTEM THERMAL LOADS

	Sensible (noncondensing heat exchanger)	Latent (condenser)
Minimum	392 Btu/hr	80 Btu/hr
Average	897	92
Active	920	201
Design Gas flow	183 lb/hr	82.5 lb/hr

During checkout of the electronics on the ground, dry nitrogen at a temperature of 70 ± 10°F will be circulated between the shroud and the spacecraft to absorb the heat dissipated by the electronics during operation.

The use of redundant pump assemblies requires that the performance of the pump in service be verified. This is accomplished by sensing the pressure rise across the pump by the pump pressure differential transducer. Should the differential pressure approach 0, indicating a loss of power, or a failure of the motor assembly, standby pump assembly is automatically enabled.

The penetration of the radiator by a meteoroid initiates an entirely different sequence of events, assuming the hole is large enough to allow all coolant to leak from the radiator in a relatively short time. As the coolant leaks from the system, the first significant bit of data processed by the pump/radiator control is the receipt of a low level alarm from the accumulator quantity transducer in the pump assembly. The second bit of informative data is the reduction in pump pressure differential which is monitored by the pump pressure differential transducer. The final data received by the pump/radiator control is that the total pressure in the coolant system, detected by the pressure sensor, is approaching zero. The coincidence of these three bits of data provide logic to the controller which initiates the following action. Power to the operating pump is interrupted to prevent possible damage during high-speed, no-load operation, or to prevent possible damage to the rotor upon refill of the coolant while the rotor is at high speed. The controller commands the composite radiator control valve to be actuated. The actuation of this valve performs the following functions simultaneously: closes the radiator coolant pass just penetrated by a meteoroid, opens the auxiliary coolant path in the radiator to coolant flow; and opens valve to allow coolant in the reservoir to flow into the coolant loop, thus recharging the coolant loop. The reservoir is pressurized from the nitrogen tank located in the ECS/WMS Subsystem by the coolant pressurization regulator/relief.

Passive thermal control subsystem: A passive system has been selected for the thermal control of the electronic equipment outside the pressurized area. Thermal control coatings, PV100 and Cat-A-Lac Black, will be used on the radiating surface wherever necessary to obtain the required heat rejection capability. Thermal insulation will be used to reduce heat losses. This system will provide maximum reliability and minimum cost and time schedule. The electronics which continuously generate heat will be attached to panels, which will in turn radiate this heat into space. The temperature of these electronics will be at 75 ± 35°F.

The thermally critical electronic equipment, illustrated in table 20, can be controlled within their temperature limits through the use of varying radiator temperatures plus thermal inertia. The temperature fluctuations created by power transients will be reduced by increasing the thermal inertia of the equipment. The television transmitter's TWT poses the most serious thermal control problem due to its low thermal inertia and high power fluctuations. It, and other such types of electronics, will be conductively mounted to the spacecraft, which will act as a thermal sink. The temperature of this electronic equipment will range from 60°F to 120°F.

Radiative thermal insulation will be used to reduce heat losses from the spacecraft and from all electronics located on top of the spacecraft. A thickness of one-half inch of insulation, aluminized mylar separated by Dimplar, will be used to limit thermal losses to 75 Btu/hr.

Advance development areas. - No special development is anticipated for the Thermal Control Subsystem, since generally its mechanization falls within the present state of the art.

Preliminary equipment list. - A preliminary list of equipment in the Thermal Control Subsystem is presented in table 21.

Structure and Mechanical

This section presents a description of the Primate Spacecraft Structure and Mechanical Subsystem derived during the study program. The overall objective of this portion of the study was to develop the Structure and Mechanical Subsystem to a level of detail sufficient to assure compliance with mission and program objectives, and to establish Phase B design criteria for the subsequent Phase 2 efforts. Consequently, the main design and analytical efforts were directed toward critical areas of the primary structure such as overall concepts, major joints and seals, and materials selections. The secondary structure and most equipment mounting provisions were examined to a lesser depth. The mechanical elements examined during the Phase 1 effort were limited to the major separation and deployment systems, the booster/spacecraft interface, the Command and Service Module/spacecraft interface, and solar panel deployment.

A structural concept consistent with the general study and program requirements was selected by a series of trade studies (ref. 6). Loads and a more detailed level of requirements were then developed with this concept and used to establish the preliminary design described in this section.

TABLE 20. - NON-CONTINUOUSLY OPERATED ELECTRONIC EQUIPMENT HEAT DISSIPATIONS FOR WORST CASE THERMAL CONTROL

Equipment	Thermal inertia Btu/°F	Continuous heat dissipation, Btu/hr	Peak heat dissipation, Btu/hr	Typical temperature limits, °F
Telemetry Transmitter	0.4	0	205* (60 watts)	10 to 130
Television Transmitter				
1. TWT	0.8	0	240* (70 watts)	10 to 130
2. Power Supply	1.6	0	240* (70 watts)	10 to 130
Batteries**				
1. Recharge	45	560 (164 watts)	560 (164 watts)	50 to 100
2. Discharge	45	70 (20.5 watts)	140*** (41 watts)	50 to 100

* Peaks occurring for 10 min/orbit/station and no peaks the remainder of the 93.75 minute orbit.

** Recharge during 57.94 minute sun exposure and discharge during 35.81 minute dark side.

*** Ten minute duration during 35.81 minute dark period

TABLE 21. - THERMAL CONTROL SYSTEM PRELIMINARY EQUIPMENT LIST

Item	Description	Suggested manufacturer	Part number	Weight per spacecraft
1	Insulating blankets	Northrop Corporation	---	85 lb
2	Paint	Vit-a-var Paint Company	PV 100	---
3	Paint	Finch Paint Company	Cat-A-Lac	---

TABLE 21. - Concluded

Item	Description	Suggested manufacturer	Part number	Weight per spacecraft
4	Tube-fin radiator	Ling-Temco-Vought Corp.	New	26 lb
5	Coolant	Minnesota Mining & Mfg. Co.	FC-75	20 lb

The mechanical elements of the subsystem did not require the synthesis and analysis of a variety of mechanisms since existing designs or hardware were readily applicable. The approach was to examine these components in sufficient depth to assure the ability of the mechanism to function in the Orbiting Primate Spacecraft and to determine the extent of rework necessary for this application.

In developing the structure and mechanical subsystems, consideration was given to other possible applications or to growth versions. One of the most likely groups of applications is that illustrated in figure 22 where the structure is depicted as a modular unit in a manner similar to the NASA MSC MMM.

Structure and mechanical subsystem requirements. - The Primate Spacecraft Structure and Mechanical Subsystem preliminary design was developed to the following requirements. The Structural and Mechanical Subsystem will support and protect the experiment and subsystem elements; and provide the mechanism necessary for separation from the launch vehicle for CMS docking, and for the deployment of such elements as solar panels and antennae.

The Structure and Mechanical Subsystem shall be compatible with both the Saturn IB and Saturn V launch vehicles; however, the SIB shall be considered as the candidate launch vehicle for the baseline spacecraft.

To facilitate low cost and early availability, the structure shall use existing and qualified concepts and elements to the extent practical and be compatible with existing SAA interfaces. The Structure and Mechanical Subsystem shall be developed to a depth compatible with confident estimates of weight, cost, and development schedules.

The following definitions were applied during the study:

(1) Criterion is a standard of judging. For example, criterion stress can mean the material yield stress, the material ultimate stress, or the component buckling load or stress, etc.

(2) Limit load is the maximum load calculated to be experienced by the structure under the specified conditions of operation.

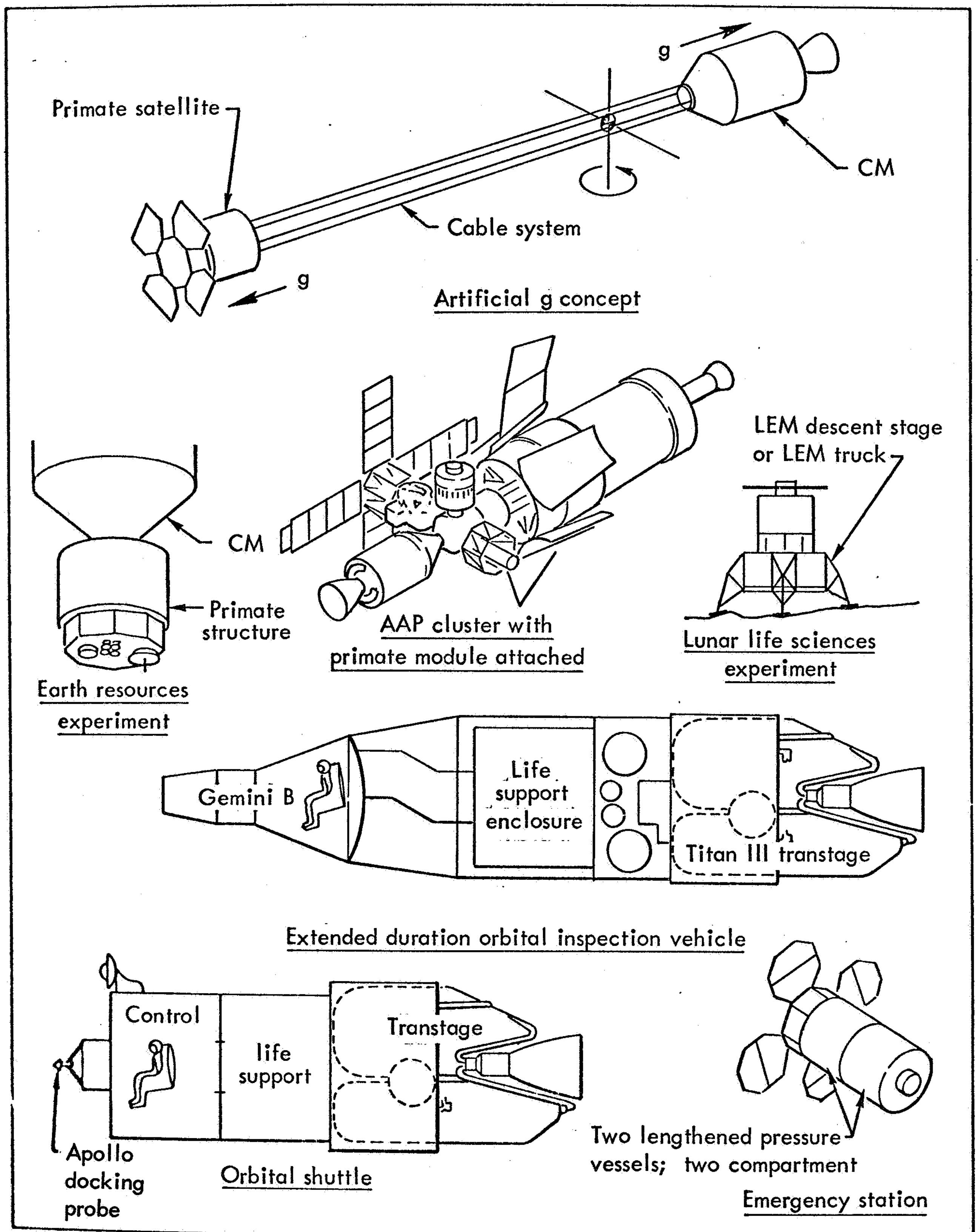


Figure 22. - Example applications of primate structure

(3) The design load is the limit load multiplied by the required minimum factor of safety.

(4) The load factor is the factor by which the steady state loads are multiplied to obtain the equivalent static effect of dynamic loads.

(5) The factor of safety is defined as the ratio of the criterion load or stress to the limit load or stress.

(6) The margin of safety is the percentage by which the criterion load or stress exceeds the design load or stress.

(7) The operating pressure is the nominal pressure to which the components are subjected under steady state conditions in service operations.

(8) The limit pressure is the maximum operating pressure or operating pressure including the effect of system environment, such as vehicle acceleration, and so forth. For hydraulic and pneumatic equipment, the limit pressure will exclude the effect of surge.

(9) Combined stresses are defined as the stresses resulting from the simultaneous action of such factors as primary stresses, secondary stresses, thermal stresses, etc.

The geometry of the Structure and Mechanical Subsystem shall be compatible with the environmental and structural concepts illustrated in the referenced figures:

(1) Launch envelope, figure 23

(2) In-orbit geometry, figure 24

(3) Spacecraft concept, figure 25

(4) Structure concept, figure 26

(5) Structure subsystem, figure 27

(6) Spacecraft Basic Dimensions set forth in Appendix A.

The weight budget, itemized in table 22, shall be used as a guide for the design of the primary structure.

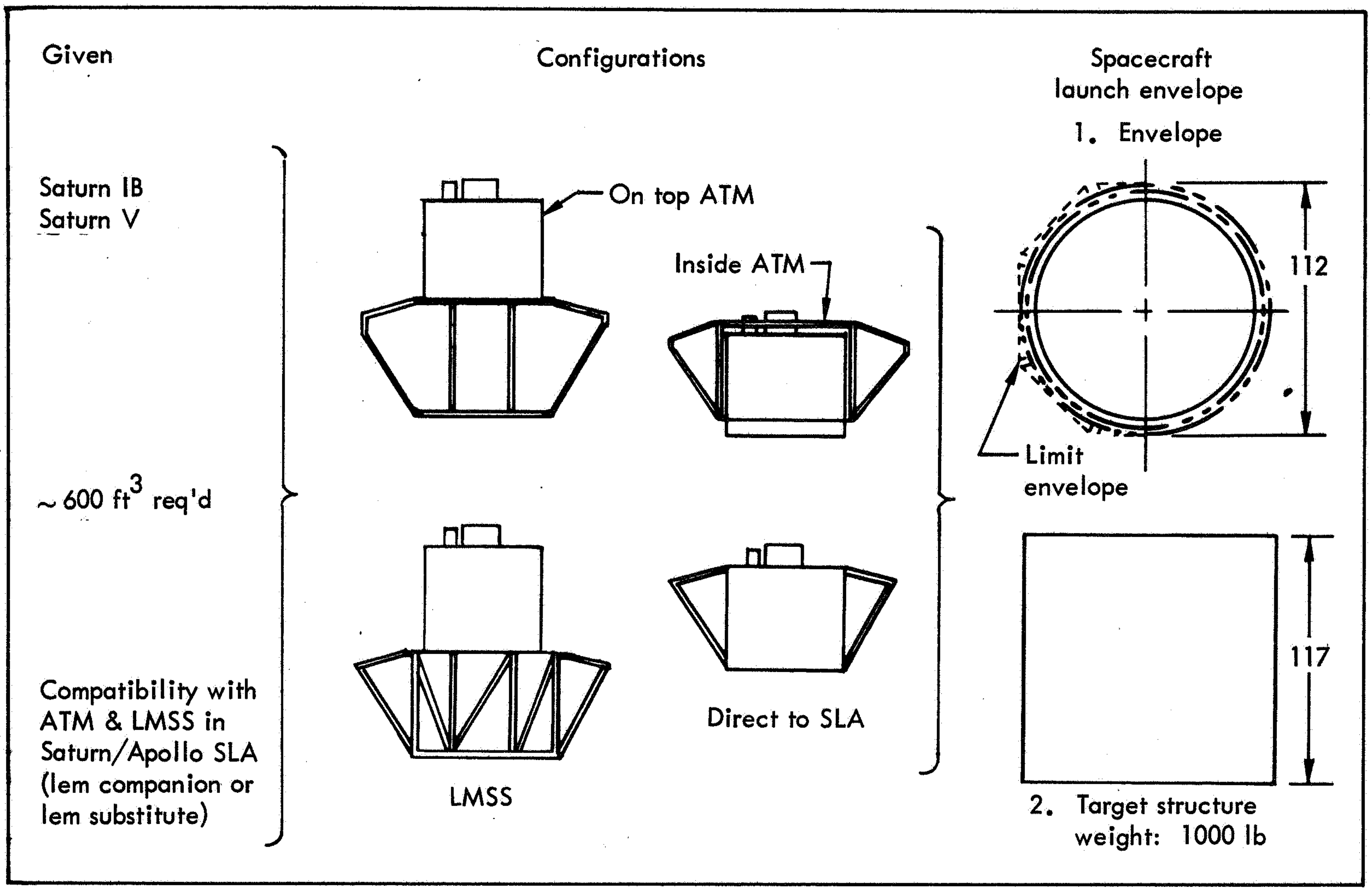


Figure 23. - Launch envelope

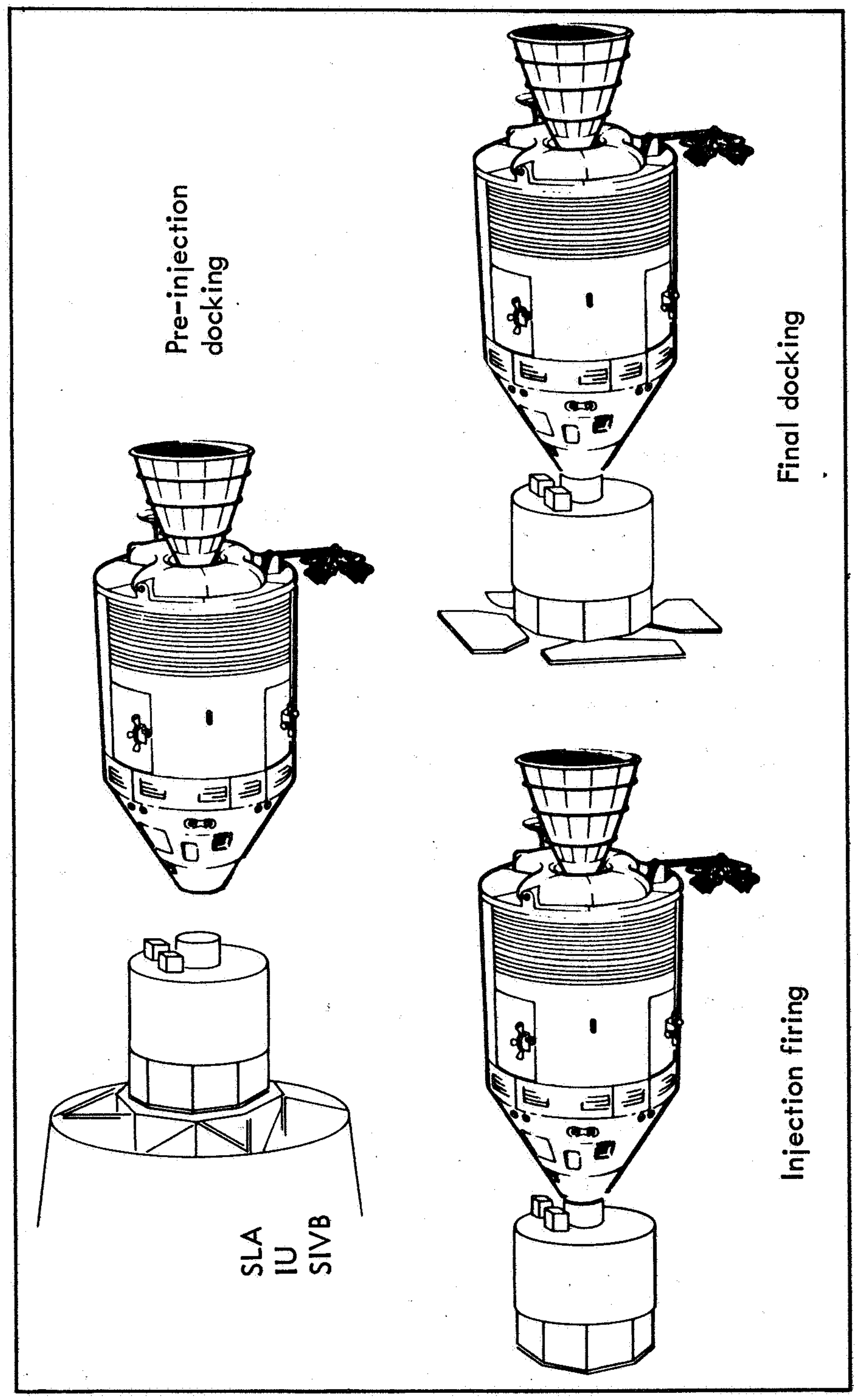


Figure 24. - In-orbit geometry

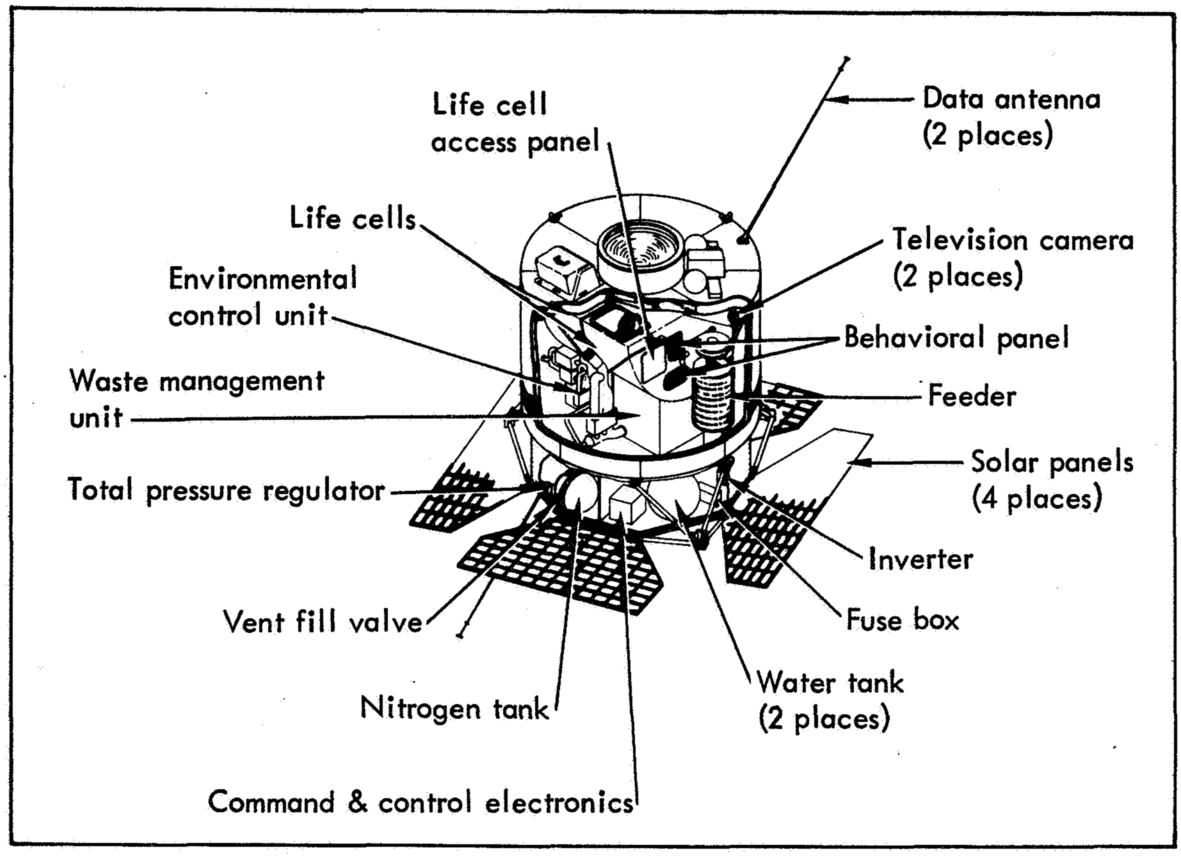


Figure 25. - Spacecraft concept

The requirements imposed on the structure due to the operating environment are listed below.

(1) Pressurized environment for the primates: 14.7 ± 2.0 p.s.i.a. The animals can tolerate 5 to 20 p.s.i.

(2) Pressurized volume for the experiment: 300 cubic feet

(3) Duration in orbit: 1 year (design point)

(4) Shielding:

(a) The maximum ionizing radiation shall be 100 rad/year maximum. Shielding requirements above that inherently provided by the structure as designed to strength and material requirements will be applied directly to the cages..

(b) **Meteroids:** P(o) ≥ 0.995 for 1 year. The structure shall be compatible with both zero g and artificial g levels from 1/6 g to 1.0 g.

(5) Materials likely to produce contaminating vapors or particulates are to be avoided.

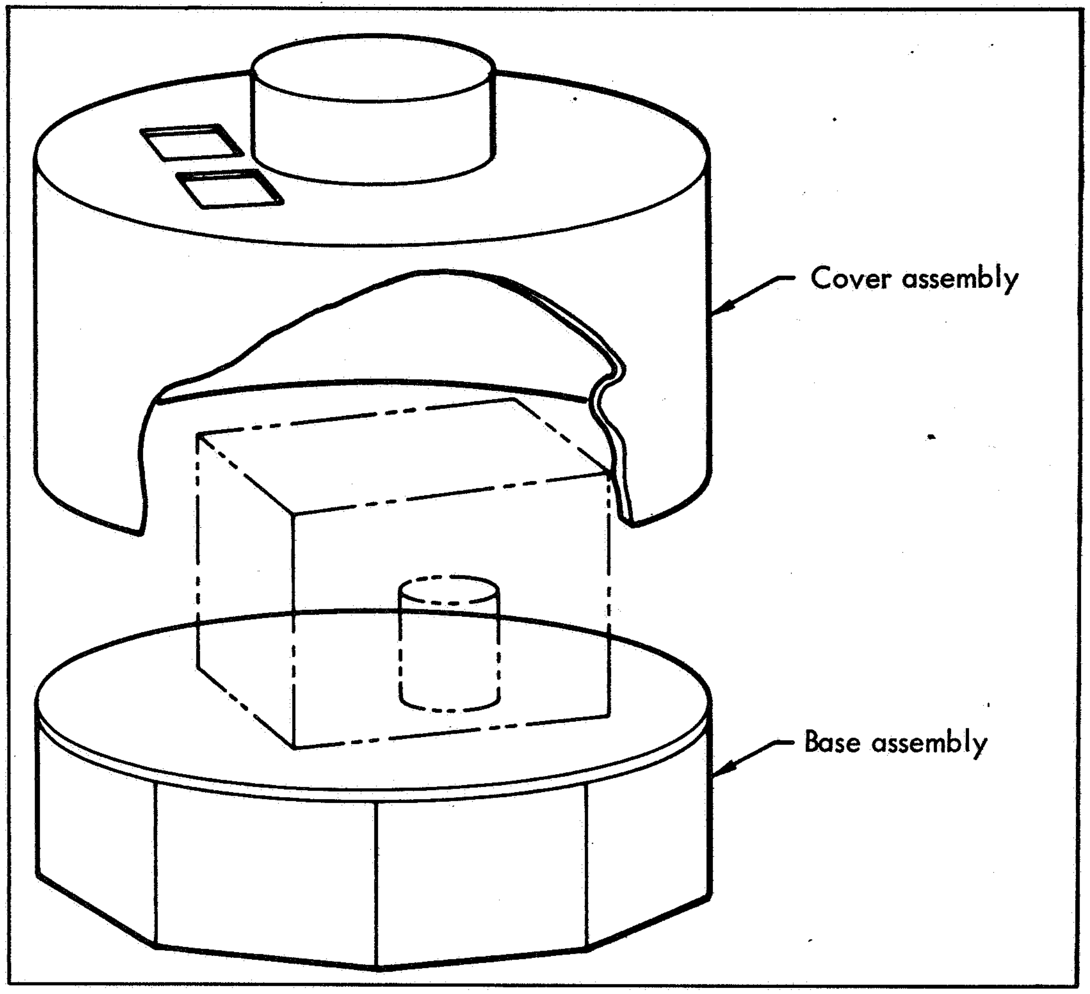


Figure 26. - Structure concept

(6) Prelaunch access shall be provided for the primates and batteries.

The structural subsystem shall be compatible with the access requirements listed in table 23.

The spacecraft will be designed to withstand the natural environments defined in the Master End Item (MEI) Specification, CP-10000 Orbiting Primate Spacecraft (ref. 7). The following is a summary of those items pertinent to the to the structural preliminary design:

(1) <u>Transportation, handling and storage</u>

(a) Temperature range: -45°F to 145°F

(b) Pressure: 3.47 psia to sea level

(c) Humidity: 0 to 100% relative humidity

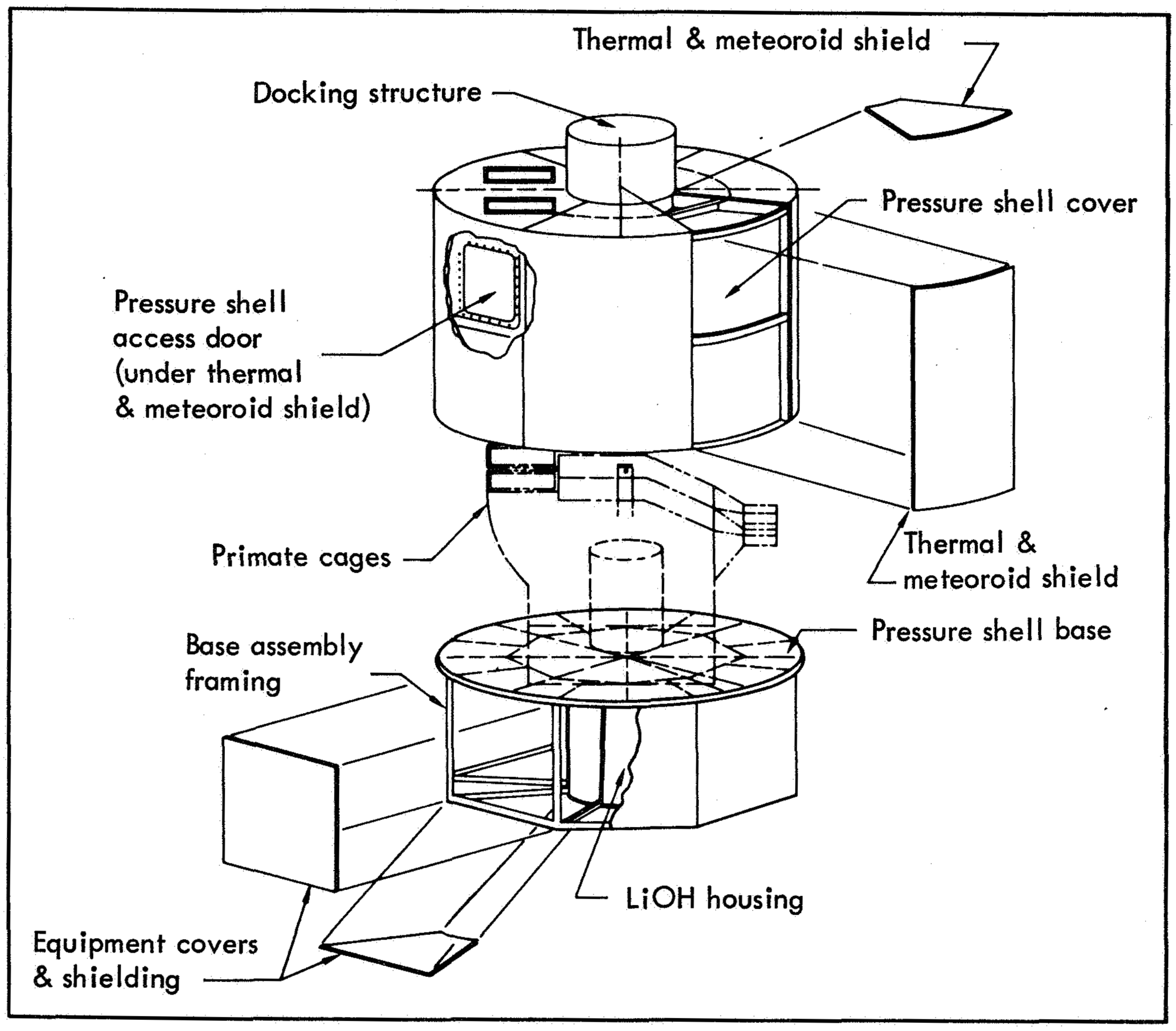


Figure 27. - Structure subsystem

(d) Sunshine, sand and dust, fungus, salt spray, and ozone as defined in the MEI specification.

(2) Prelaunch

(a) Temperature: 70°F ± 30°F

(b) Pressure: sea level

(c) Relative humidity: 50 ± 10%

(d) Ground winds: Per KSC wind data at height intervals of 10 to 400 feet.

TABLE 22. - PRIMARY SPACECRAFT WEIGHT BUDGET

Item	Target	Allowable
Pressure Shell Cover Assembly	500	750
Base Structure Assembly	400	550
Miscellaneous fasteners, etc.	20	25
Contingency	50	75
Total	970	1400

TABLE 23. - SPACECRAFT ACCESS REQUIREMENTS

Period	Section	
	Pressurized	Non-pressurized
Manufacture and assembly	All items accessible for assembly, test and checkout.	
Launch preparation (before pressure shell main seal)	All items for remove and replace, test and checkout, etc.	
Launch preparation (after pressure shell main seal)	Recovery capsule access, TV, ECS, feeders, etc.	All unpressurized items
Prelaunch	TV, ECS, feeders, primates, etc.	Tankage, electrical and electronic equipment, RCS, etc.
EVA in orbit	None	Release and deployment mechanisms and recovery capsules

(3) Orbit

(a) Thermal radiation: (see ref. 7)

(b) Pressure: 10^{-13}mm of Hg

(c) Meteroid: (ref. 7 and 14)

(d) Ionizing radiation: (see ref. 7)

(4) Recovery

CM interior: (see ref. 7)

The stress in any element of the structure will not exceed the allowable stress of the material when subjected to ultimate loads. The allowable strength of the material will in general be taken from MIL Handbook 5. The minimum guaranteed values will be used. However, in cases where there is no pronounced yield point, the yield point will be taken as the 0.2 percent offset value. For materials where the yield point cannot be established, the safety factor against ultimate will govern. Strength data will include such effects as temperature, radiation, fatigue, impact, and creep.

A factor of safety of 1.5 will be generally used for all inertia-type loads but may be reduced to 1.35 for special cases. The prelaunch and launch loads (i.e., prior to q max) will use a load factor of 1.15 to account for dynamic effects. Exit flight loads (i.e., after q max) and orbit maneuver loads will use a factor of 1.2 to account for dynamic effects.

The pressure vessel will be designed to withstand a yield pressure of 1.4 times the limit pressure, an ultimate pressure of 2.0 times the limit pressure, and a proof pressure of 1.33 times the limit pressure. The ultimate pressure combined with flight loads will be 1.5 times the limit pressure. High pressure tanks and accumulators will be designed to withstand a proof pressure of 2.0 times maximum operating pressure and an ultimate pressure of 4.0 times maximum operating pressure. The liquid storage tanks will be designed to withstand a proof pressure of 1.4 times maximum operating pressure and an ultimate pressure of 2.0 times maximum operating pressure.

Ground Handling loads shall not determine primary structure design. However, a vertical load factor of 2.0 will be applied for hoisting and lifting. Hoisting and lifting forces shall be applied within a cone angle of 10° from the vertical through the pick-up points.

The structure will be designed to withstand the launch and boost loads given in table 24 for the reaction and attach point geometry defined by figures 28, 29, and 30. Reaction equations are given in Appendix B. The spacecraft will be designed to withstand the transposition docking loads shown in figure 31. The spacecraft will be designed to withstand the loads imposed by the CSM SPS during orbit transfer; these loads are shown in table 25. The derivation of the loads is given in Appendix C.

The pressure vessel, will be designed to withstand a limit pressure of 16.7 psia, and the docking collar will be designed to withstand the loads imposed by the command module internal pressure as shown in figure 32.

TABLE 24. - ATTACH POINT LIMIT LOADS

Attach points	Condition	Design Point Limit Loads (lbs) X	Y	Z
A & B	1	3500	±300	±500
A & B	2	±1125	±250	±440
A & B	3	±1000	±360	0
A & B	4	±660	0	±620
C & D	5	8000	±200	±125
C & D	6	6700	±700	±500
C & D	7	±600	±900	0
C & D	8	±660	0	±620
C & D	9	±900	620	±440

Structure and mechanical subsystem. - This section presents a description of the spacecraft primary structure and the principal mechanical devices. Secondary structural elements, such as equipment mounting bracketry, are treated only briefly since difinition of their characteristics is not critical to the preliminary design and since their specific requirements will largely be determined by the final selection and arrangement of equipment items during the Phase 2 effort. The preliminary stress analysis performed in developing the baseline design is presented in Baseline Spacecraft Preliminary Stress Analysis (ref. 12).

The general arrangement of the structure is shown in figure 33. An exploded view of the components and subassemblies is depicted in figure 34.

The structure consists of two major elements: the base assembly and the pressure shell assembly. Assembled and sealed, they provide the pressure vessel necessary to contain the 14.7 ± 2.0 psi environment for the experiment, the mechanical interfaces for the launch vehicle and the Command/Service Module, and protection for the various items of equipment comprising the remaining subsystems.

Conventional aerospace construction is used throughout. Selected alternatives were based on the best choice relative to compatibility with mission requirements, availability, materials, manufacturing capabilities, re liability, cost, and flexibility. For example, machined and welded flat bulkheads were selected over domed ends for simplicity in manufacture and equipment installation even though this alternative is accompanied by a 50

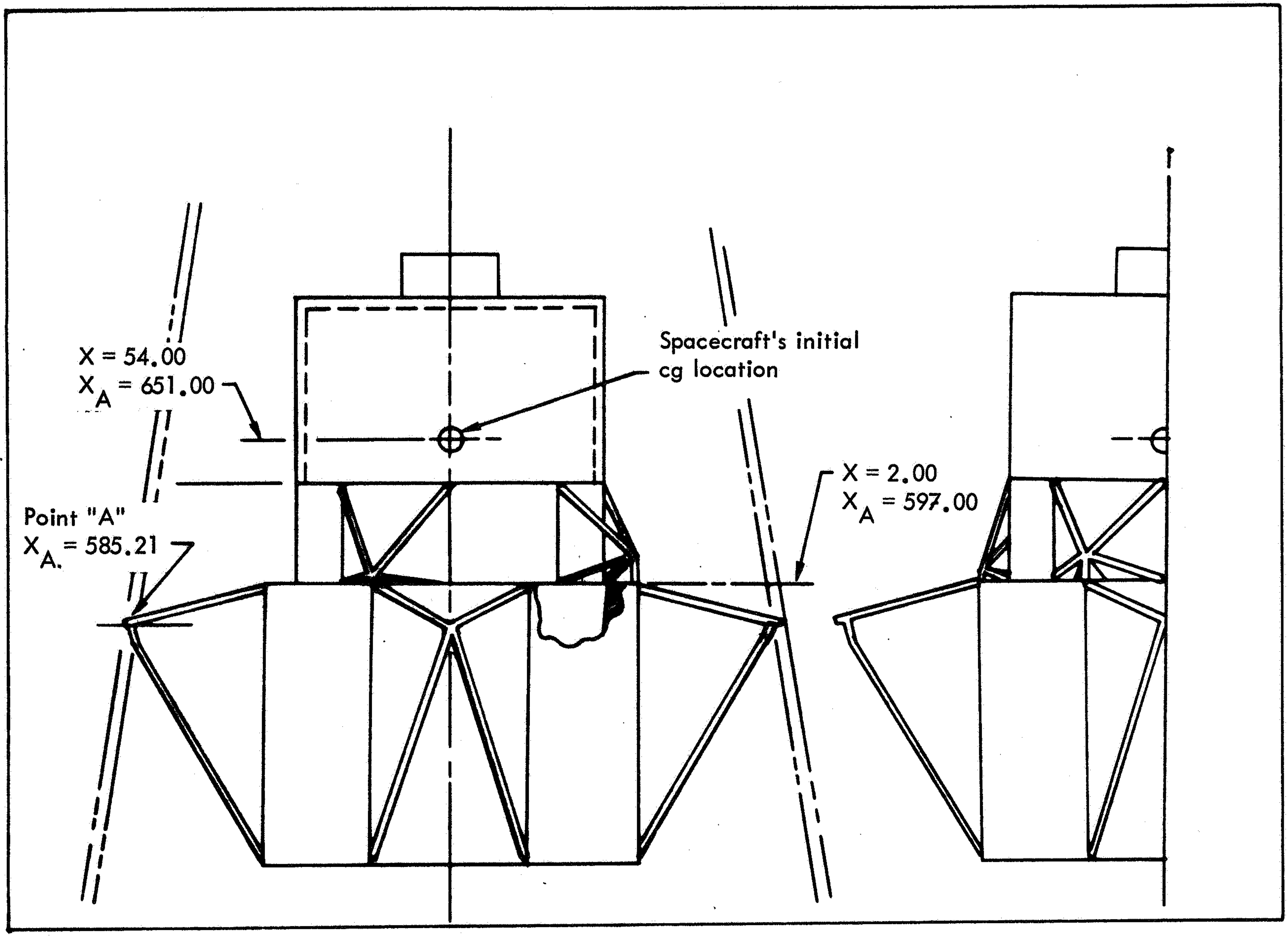


Figure 28. - Primate spacecraft design point installation (ref MSFC dwg SK30-3812 DTD 2-15-67)

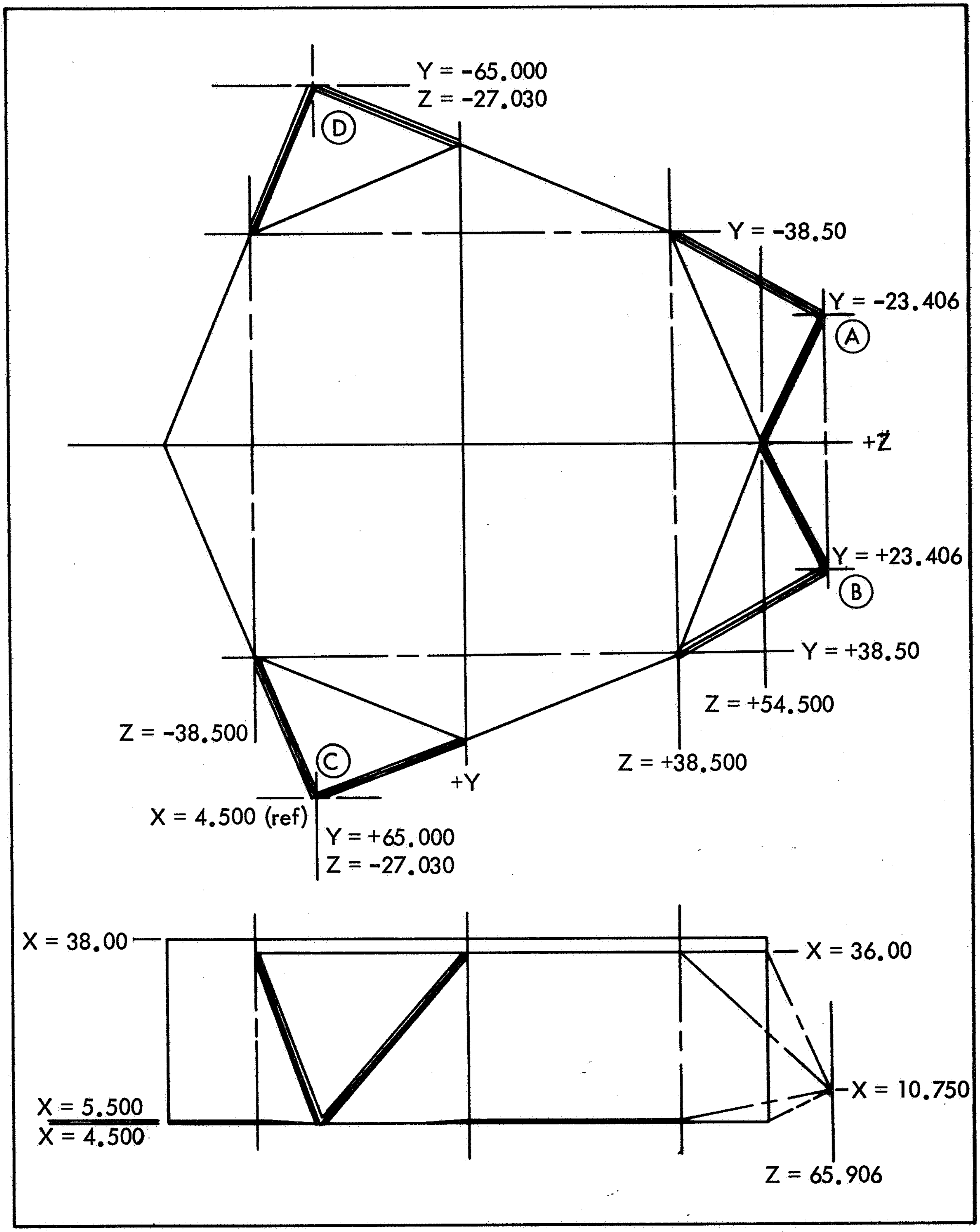
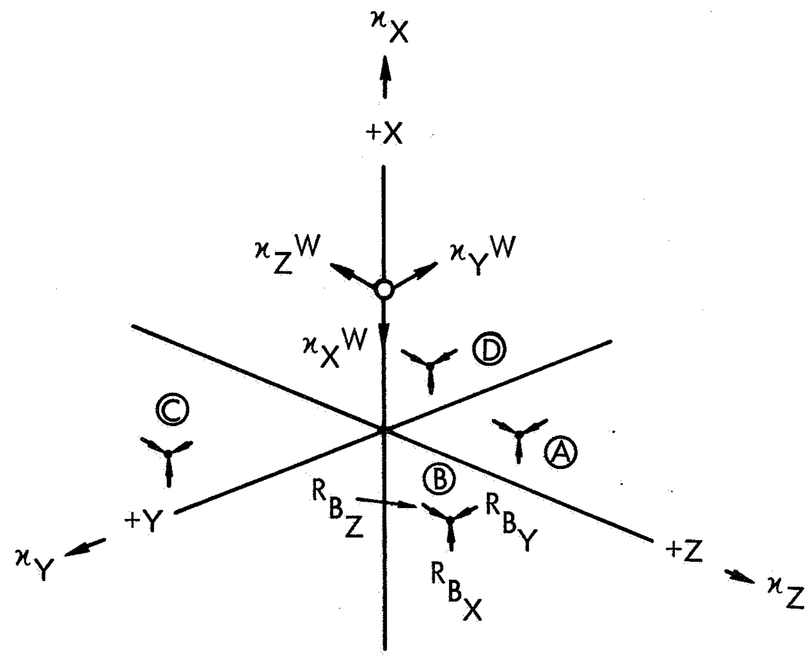


Figure 29. - Primate S/C to ATM attach point geometry



Reactions due to $\varkappa_X$

$$R_{A_X} = 0.1455\,\varkappa_X W$$
$$R_{B_X} = 0.1455\,\varkappa_X W$$
$$R_{C_X} = 0.3545\,\varkappa_X W$$
$$R_{D_X} = 0.3545\,\varkappa_X W$$

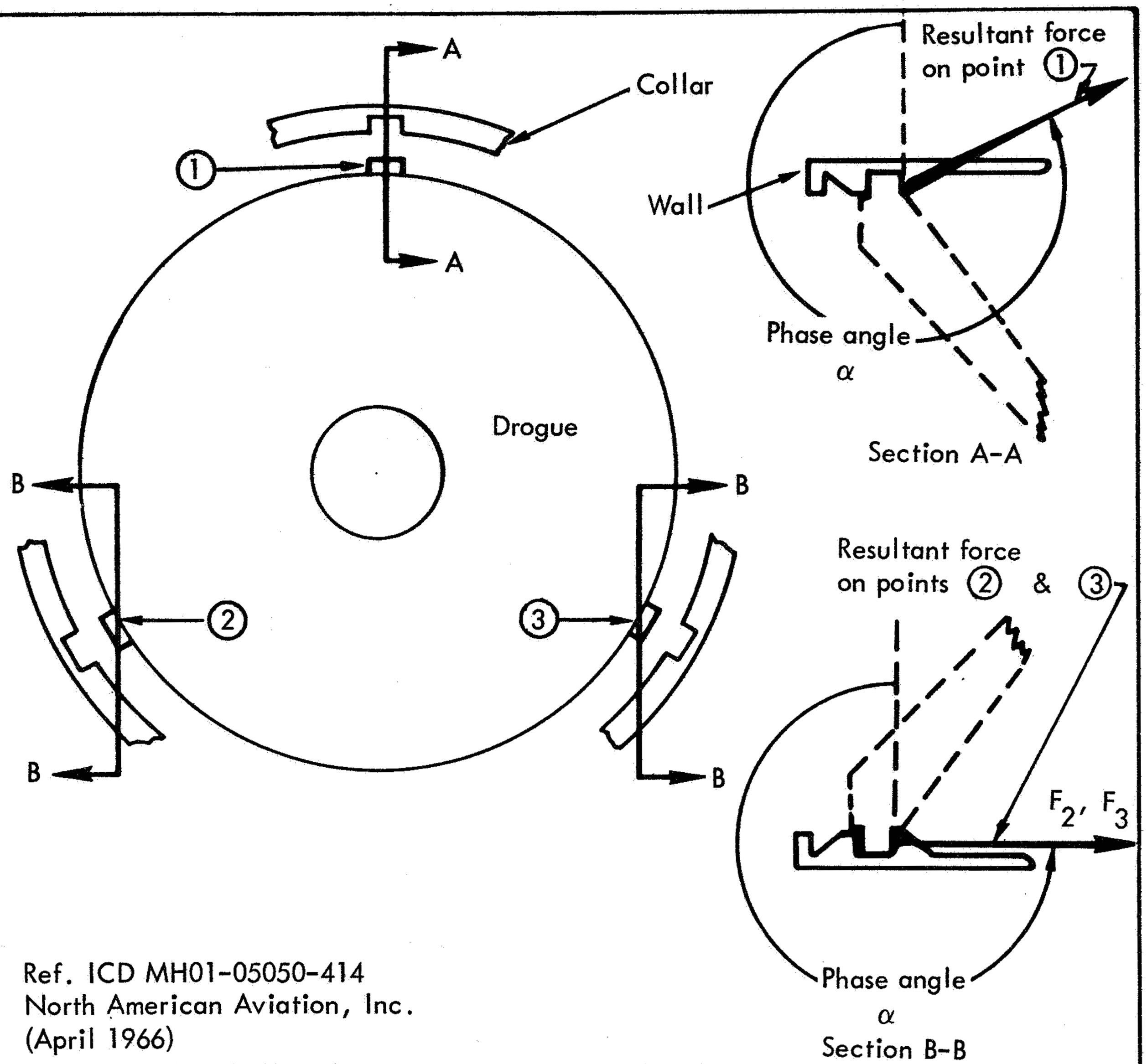
Reactions due to $\varkappa_Y$

$$R_{A_X} = +0.381\,\varkappa_Y W$$
$$R_{B_X} = -0.381\,\varkappa_Y W$$
$$R_{C_X} = -0.2435\,\varkappa_Y W$$
$$R_{D_X} = +0.2435\,\varkappa_Y W$$
$$R_{A_Y} = +0.1455\,\varkappa_Y W$$
$$R_{B_Y} = +0.1455\,\varkappa_Y W$$
$$R_{C_Y} = +0.3545\,\varkappa_Y W$$
$$R_{D_Y} = +0.3545\,\varkappa_Y W$$

Reactions due to $\varkappa_Z$

$$R_{A_X} = -0.266\,\varkappa_Z W$$
$$R_{B_X} = -0.266\,\varkappa_Z W$$
$$R_{C_X} = +0.266\,\varkappa_Z W$$
$$R_{D_X} = +0.266\,\varkappa_Z W$$
$$R_{A_Z} = +0.25\,\varkappa_Z W$$
$$R_{B_Z} = +0.25\,\varkappa_Z W$$
$$R_{C_Z} = +0.25\,\varkappa_Z W$$
$$R_{D_Z} = +0.25\,\varkappa_Z W$$

Figure 30. - Attach point reactions



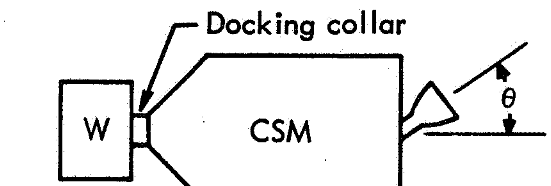
Ref. ICD MH01-05050-414
North American Aviation, Inc.
(April 1966)

Time (sec. from impact)	Point 1	Point 2	Point 3
t = 0.0	F = 3400 lb $\alpha = 320^\circ$	F = 210 LB $\alpha = 270^\circ$	F = 210 lb $\alpha = 270^\circ$
t = 1.35	F = 900 lb $\alpha = 270^\circ$	F = 840 lb $\alpha = 270^\circ$	F = 840 lb $\alpha = 270^\circ$
t = 2.0	F = 1300 lb $\alpha = 320^\circ$	F = 100 lb $\alpha = 270^\circ$	F = 100° $\alpha = 270$

Figure 31. - Transposition docking loads

TABLE 25. - ORBIT TRANSFER LOADS (SPS IMPOSED)

Condition	P_1 (lb)	P_2 (lb)	M (lb-in)
1 CSM initial weight, $\theta = 0$	3630	-	-
2 CSM final weight, $\theta = 0$	3900	-	-
3 CSM initial weight, $\theta = 8.5^{\circ}$	5620	865	68,700
4 CSM final weight, $\theta = 8.5^{\circ}$	6130	966	77,200



Conditions

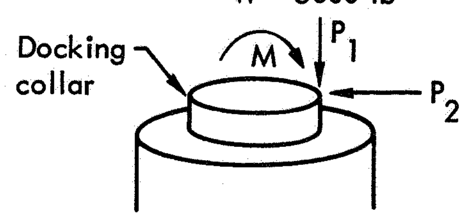
CSM initial weight = 25,117 lb

CSM final weight = 23,100 lb

SPS thrust = 21,900 lb

$\theta_{max} = 8.5^{\circ}$

W = 5000 lb



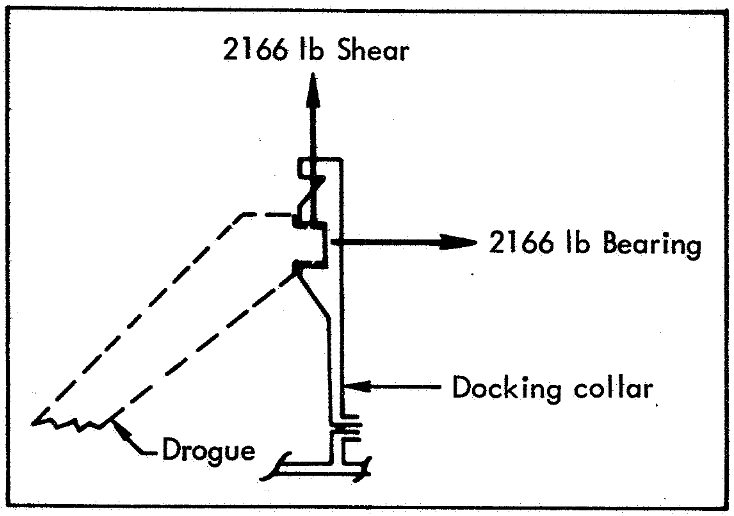


Figure 32. - Pre-separation pressure loads

to 100 pound weight penalty. Conventional materials, mainly aluminum alloys, are used throughout since extensive statistical information exists in applications similar the Primate Spacecraft.

The majority of subsystem elements for such functions as life support, thermal control, power, telecommunications, and command and control, are installed on the base structure. This arrangement operationally simplifies the final closing of the pressure shell since nearly all of the assembly and checkout operations can be conveniently performed prior to the final assembly and sealing of the base and shell. This arrangement also tends to minimize the problem of pressure shell penetration and the number of plumbing and cabling connections to be performed after sealing. Further, this arrangement can tolerate a significant degree of change in installed equipment with little or no effect on the primary structure.

Prior to assembling the pressure shell and base, all subsystem elements are readily accessible. After assembly, access to the interior of the pressure shell is restricted to that provided by access doors in the side of the cylinder and the recovery capsule openings in the upper bulkhead. The access doors are stressed skin elements and are installed and sealed during the final phases of the prelaunch activities; the recovery capsule openings are closed by the installation of the recovery capsules. The equipment items installed in the unpressurized area are accessible by means of the side panels which may be completely removed or hinged to expose the various equipment items. The volume between the pressure wall and external skin is also readily accessible by removing the appropriate panels, all of which are non-load-carrying structures.

The pressurized structure is designed to comply with the pressure and inertial loads, and the meteroid shielding requirements only. No additional mass has been added for radiation shielding. The hazards of meteroid puncture are circumvented by the classical bumper shield as shown in figure 33, the only unshielded area being the end of the lithium hydroxide can which

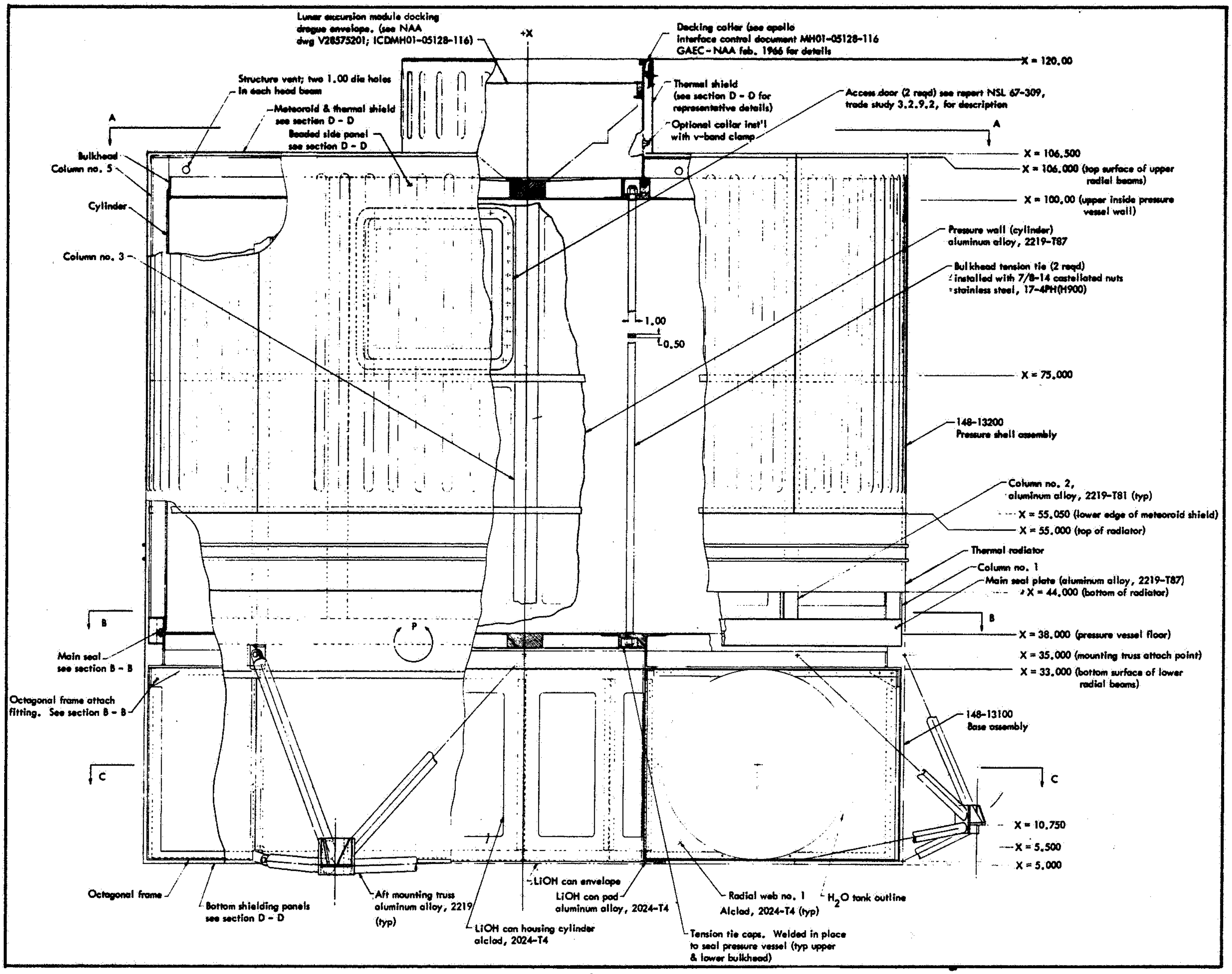


Figure 33. - General arrangement drawing

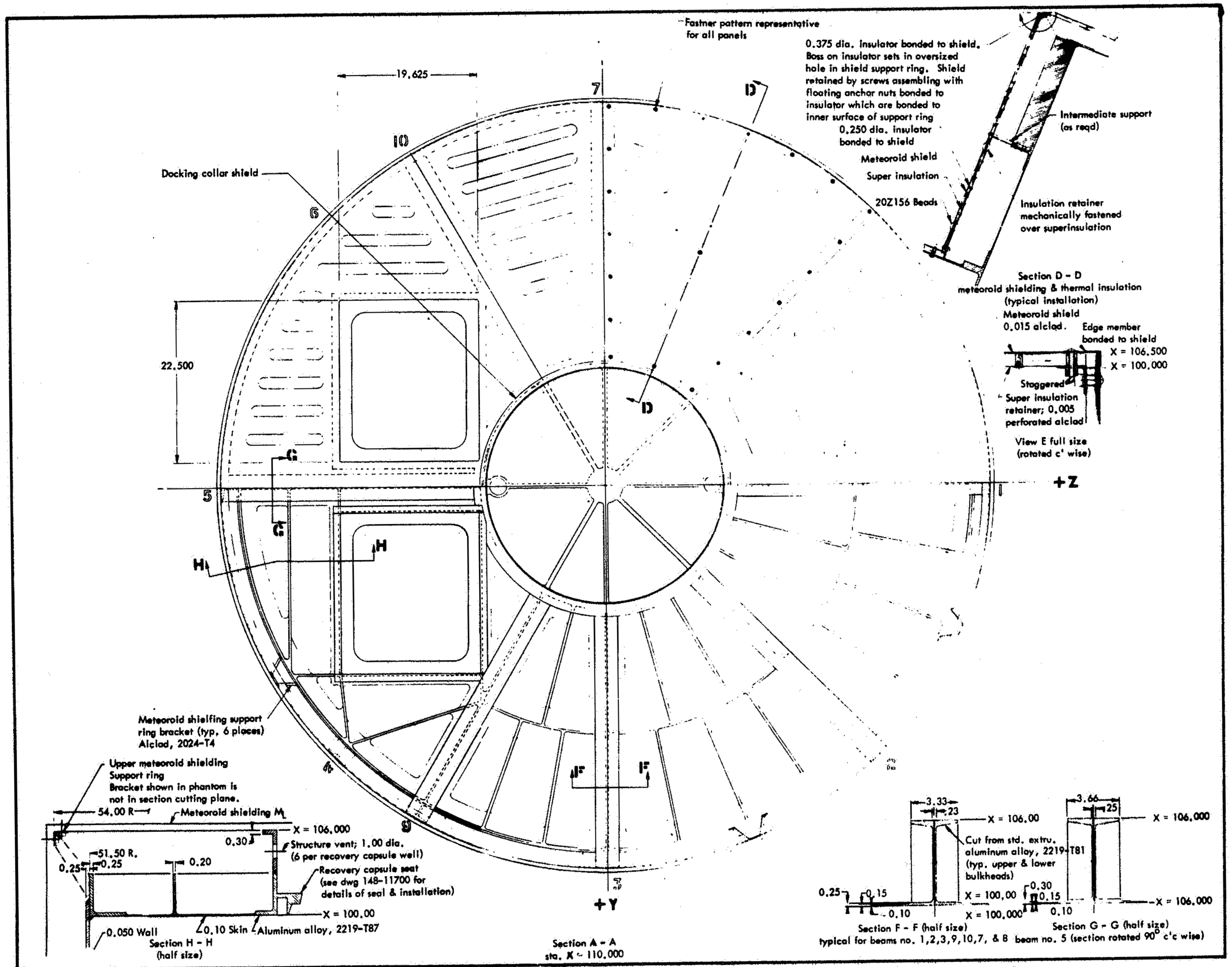


Figure 33. - General arrangement drawing (continued)

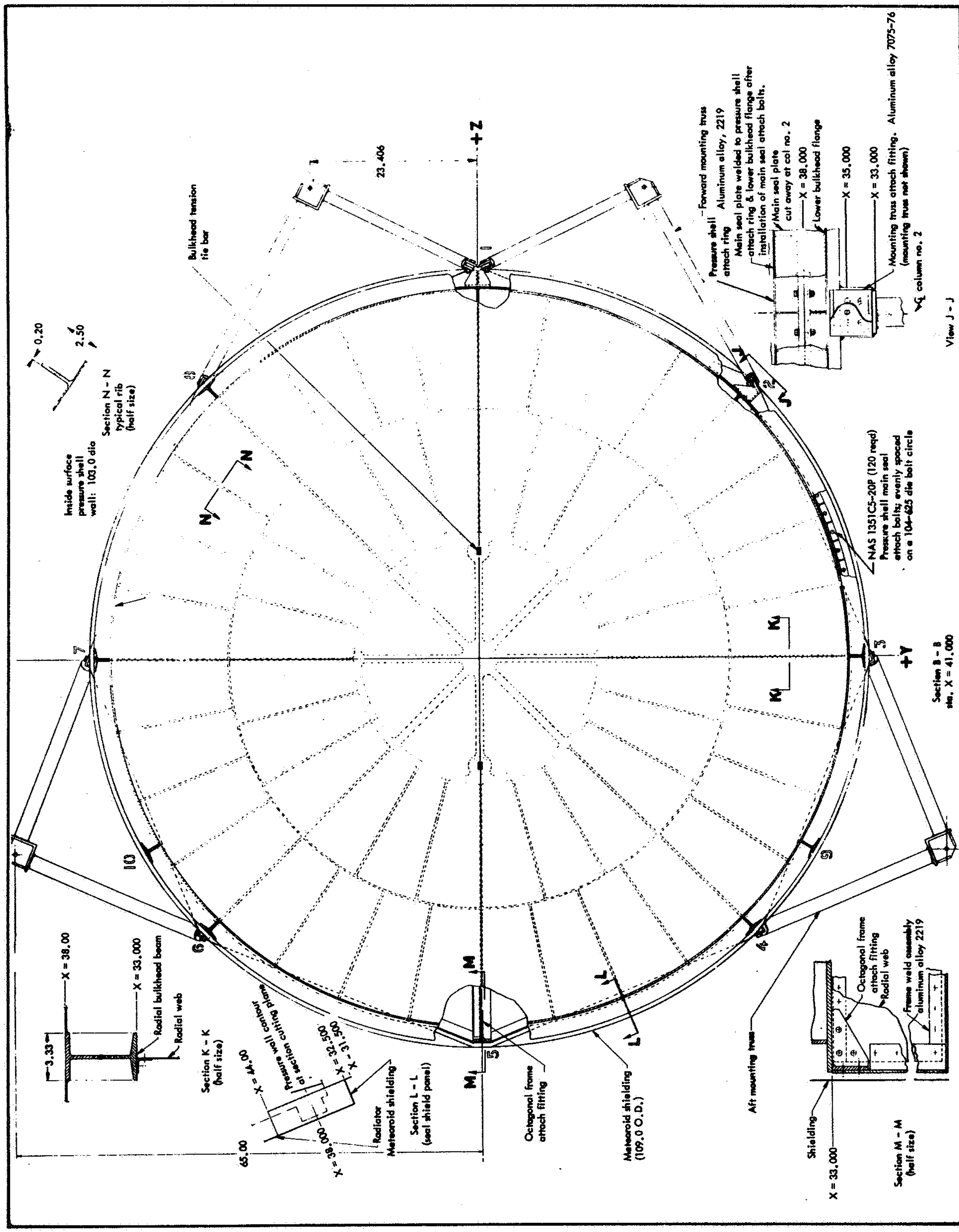


Figure 33. - general arrangement drawing (continued)

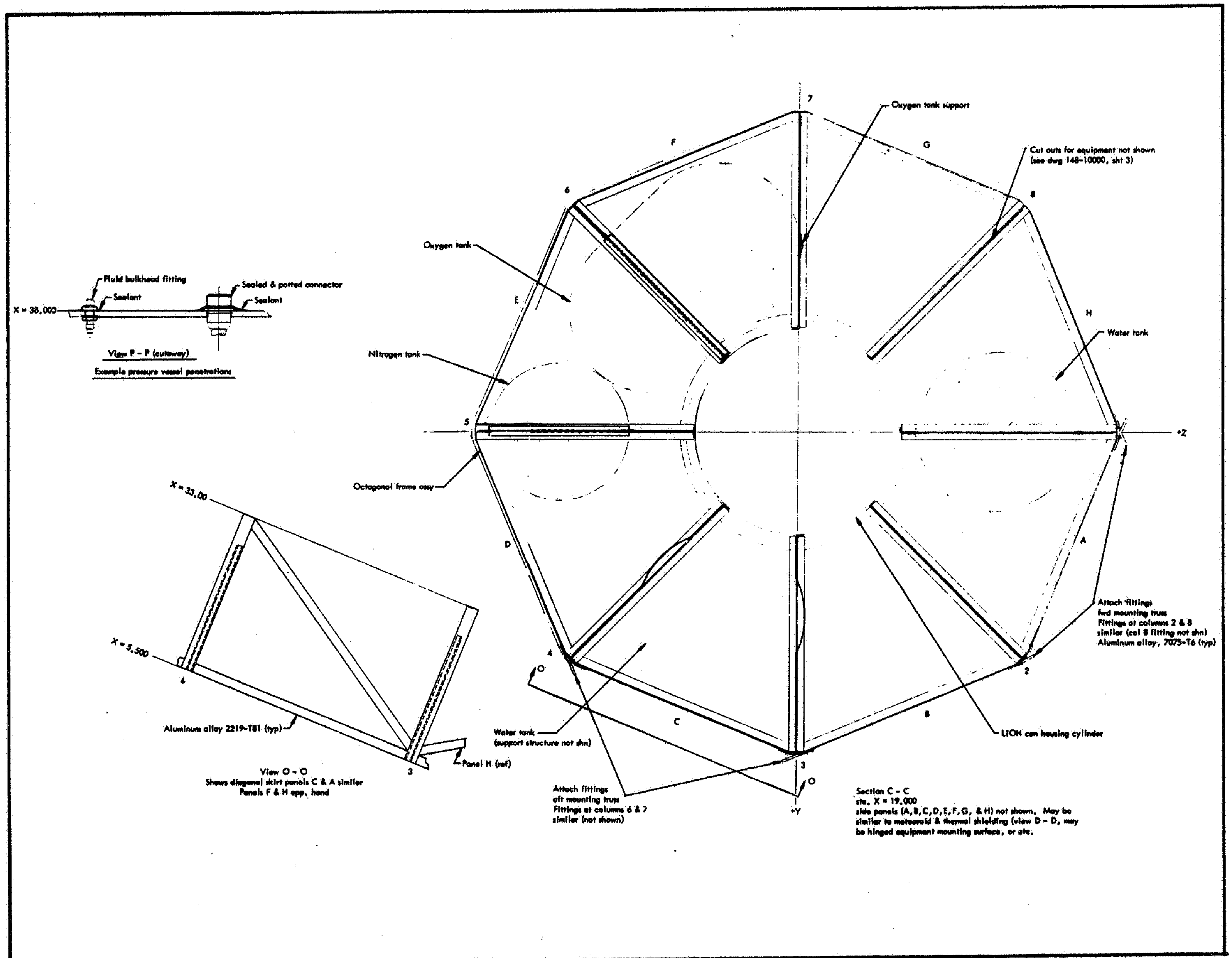


Figure 33. - General arrangement drawing (concluded)

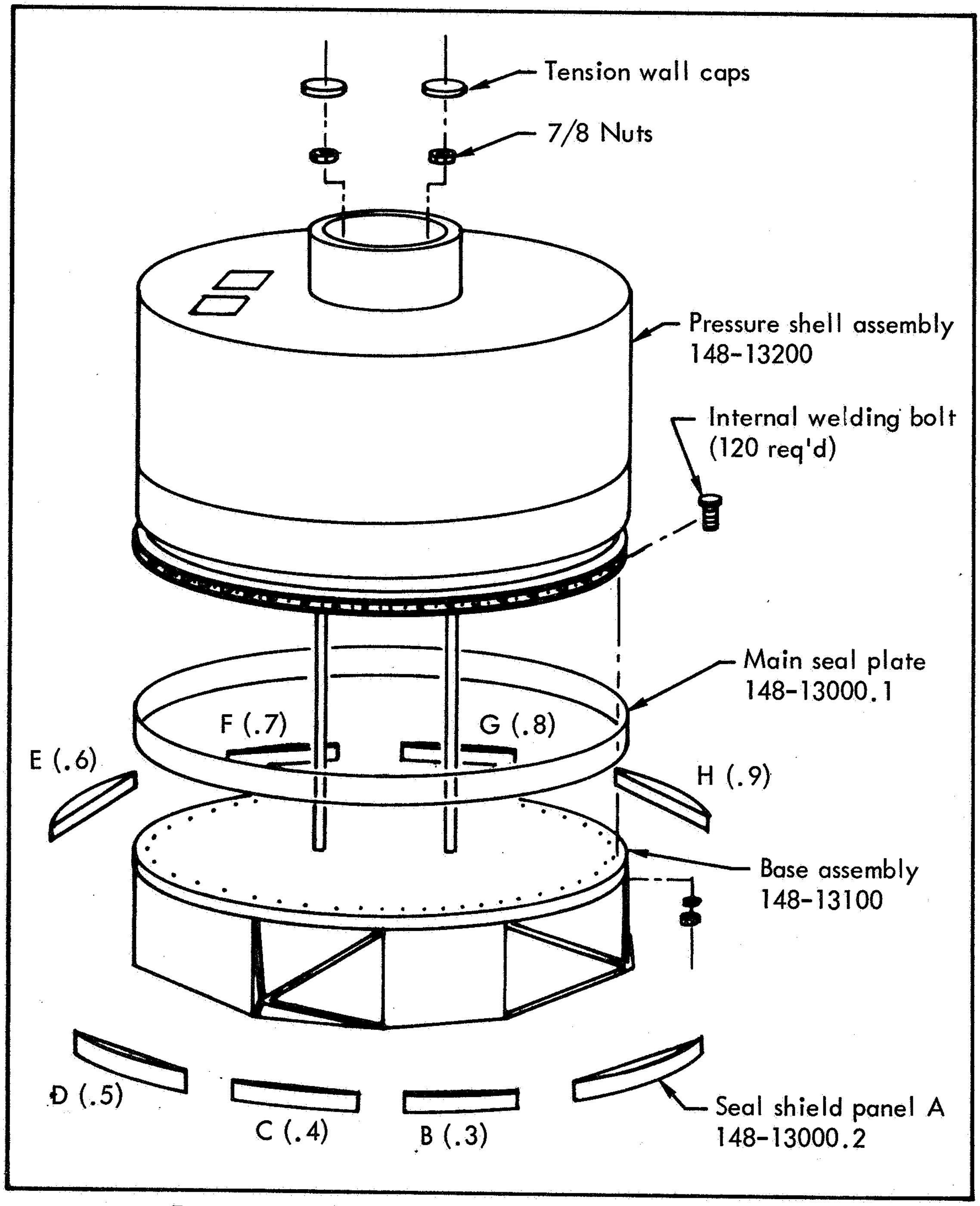


Figure 34. - Exploded view of structure subsystem

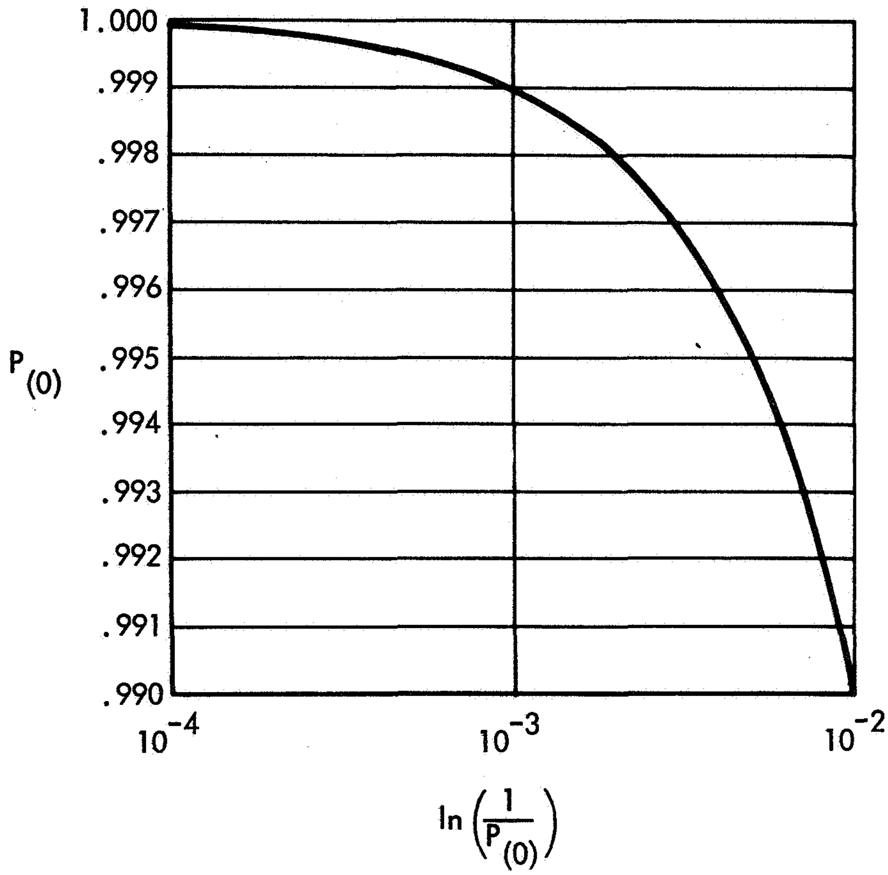
is treated as single sheet armor. Calculations using the relationship $P_{(0)} = e^{(-4AT\eta^{-1} t^{-3} \times 10^{-10})}$ where $P_{(0)}$ is the probability of no penetration, T is the ft^2-days parameter, η is the mutliple skin efficiency factor, and t is the effective thickness, yield a probability of no meteroid pressure vessel penetration for 1 year of 0.9982. An analysis using the penetration frequency data presented in NASA TN D-3717 (ref. 14) indicates a rate of 0.10643 penetrations annually, or one penetration each 9.4 years. The penetration analysis is summarized in table 26. Figure 35 was used to graphically solve for $P(0)$.

The weight of the structure is estimated to be 1,394 pounds as shown in table 27. This value can be decreased by any of the following alternatives or their combinations.

The design pressure to which the structure was designed, 33.4 psia, is determined by applying a factor of safety of 2.0 to the limit pressure, 14.7 ± 2.0 psia. This is the safety factor generally used for manned spacecraft and was used as a conservative requirement for the structure. It may not be necessary to apply this same criteria, however, since the pressure vessel is not intended for human occupancy, and any hazard due to over pressure can be countered by the use of relief values, blow-out plugs, or burst diaphrams, together with suitable ducts and orifices to balance any reaction torques. With this latter consideration, a less conservative but still acceptable safety factor might be used. For example, a value of 1.5 applied to the 16.7 psia limit pressure results in a design pressure of 25 psia. Designed to this lower pressure, the weight of the structure is reduced to about 1,200 pounds, a decrease of 15 percent. Figure 36 illustrates this weight trend as a function of design pressure and factor of safety.

Reduced inertial loads can reduce the weight of the structural subsystem; however, this is an unlikely approach for the primate spacecraft for two reasons: first the critical inertial loads are established mainly by the launch vehicle and its flight profile and any significant reduction in the load factors would probably result in a substantial reduction in payload capability due to such factors as gravity losses. Secondly, reduction of weight would be reflected in only about 10% of the structure mass.

The structural weight may also be reduced by using advanced materials such as alloys of beryllium, titanium, or the 7002 or 7106 series aluminums. The beryllium-aluminum alloys, for example, such as Lockalloy, appear especially attractive for the pressure vessel structure. Lockalloy has tensile ultimate and yield strengths comparable with the 2219 aluminum as used in the baseline design, weighs about 75% as much as aluminum, is weldable, and has better modulus/density and yield strength/density ratios than the aluminum alloys,(see table 28).Extensive use of this material could result in a structure weight of approximately 1,100 pounds - about a 20% reduction over aluminum. Consideration of cost and the availability of sizes needed for the Primate Spacecraft, however, together with flight experience in aerospace applications and consideration of safety, - beryllium oxide is toxic - suggest that weight reduction through the use of advanced materials is of limited value for this program unless payload constraints become significantly more severe.



$$P_{(0)} = e^{(-4AT\varkappa^{-1}\, t^{-3} \times 10^{-10})}$$

$$\frac{1}{P_{(0)}} = e^{\left(\frac{4AT \times 10^{-10}}{\varkappa t^3}\right)}$$

$$\ln\left[\frac{1}{P_{(0)}}\right] = \frac{4AT \times 10^{-10}}{\varkappa t^3}$$

Figure 35. - Penetration probability relationship $P_{(0)} = f\left(\ln\left[\frac{1}{P_{(0)}}\right]\right)$

TABLE 26. - SUMMARY, METEOROID PENETRATION ANALYSIS

Area	Location	Skin thickness (in.) Pressure wall (a)	Bumper	Efficiency factor	Effective thickness (inches)	Area (ft^2)	AT (360A) ft^2 -days	$P_{(0)}$ (Graphically) see figure 34 (b)	Annual penetrations (c)
A_1	Upper bulkhead	0.10	0.02	0.29	0.414	58.0	20,900	0.9999	.00200
A_2	Cylinder side	0.05	0.02	0.29	0.241	140.0	50,400	0.99855	.01080
A_3	Lower bulkhead	0.10	0.02	0.29	0.414	51.2	18,420	0.9999	.00180
A_4	LiOH can side	0.10	0.02	0.29	0.414	24.0	8,640	0.9999	.00083
A_5	LiOH can end	0.10	---	1.0 (single sheet)	0.345	6.7	2,410	0.99995	.09100
Σ						279.9			.10643
$P_{(0)}$ overall								0.9982	

(a) These values are conservative; actual thickness varies from 0.1 to 0.25 and includes, for example, 6" x 3" I beams and 0.25 x 2.9 ribs.

(b) Data in this column calculated from $P_{(0)} = e^{(4AT\eta^{-1}t^{-3} \times 10^{-10})}$

(c) Data in this column derived from NASA TN D-3717

TABLE 27. - STRUCTURE AND MECHANICAL SUBSYSTEM WEIGHT

Item	Unit weight	Quantity per spacecraft	Total weight (lb)
Structure & mechanical subsystem	-	1	1468.0
Structure subsystem, complete	1394	1	1394.0
Base assembly	562	1	562
Lower bulkhead assembly	303		
Octagonal frame assembly	128		
Mounting trusses & fittings	41.5		
Side & bottom panels	50.4		
Tension tie	19.6		
Secondary structure, fasteners, etc.	19.2		
Pressure shell assembly	756	1	756
Pressure shell weld assembly (upper bulkhead, cylinder etc.)	674		
Meteroid shielding & support structure	56.5		
Access doors	4.0		
Secondary structure, fasteners, etc.	21.0		
Main seal plate, tension tie well caps, etc.	4.6		4.6
Seal shielding panels	1.4	8	11.2
Fasteners, etc.	10.0		10.0
Contingency	50.0		50.0
Mechanical subsystem complete	72	1	72

TABLE 27. - (concluded)

Item	Unit weight	Quantity per spacecraft	Total weight (lb)
Separation & deployment	4.0	1	4.0
Docking drogue	40.00*	1	40.0
Solar panel deployment mechanism			12.0
Deployment actuation mechanism	0.32	8	2.56
Pin puller	0.36	1	0.36
Outer panel release mechanism	2.25	1	2.25
Lunar panel release mechanism	2.25	1	2.25
Boost dampers	0.1	20	2.00
Cruise dampers & latch	0.5	8	4.0
Omnidirectional antenna deployment mechanism	1.37	2	2.75
Fasteners & mounting brackets, etc.			5.0
Contingency		—	10.0

* Budget

Structural weight of the baseline spacecraft may also be reduced by 100 to 150 pounds by replacing the flat bulkheads with a domed configuration. As noted in subsystem trade study (ref.6), this is accompanied by an increased manufacturing cost, reduction in the internal volume of the pressure vessel and a reduction in allowable height for the life cell cage and waste disposal unit. The net effect yields a configuration less compatible with mission requirements.

Structural weight may also be reduced by reducing the size of the pressure vessel, especially the diameter. This however, would require smaller cages and would tend to cramp the installation in the pressurized invironment.

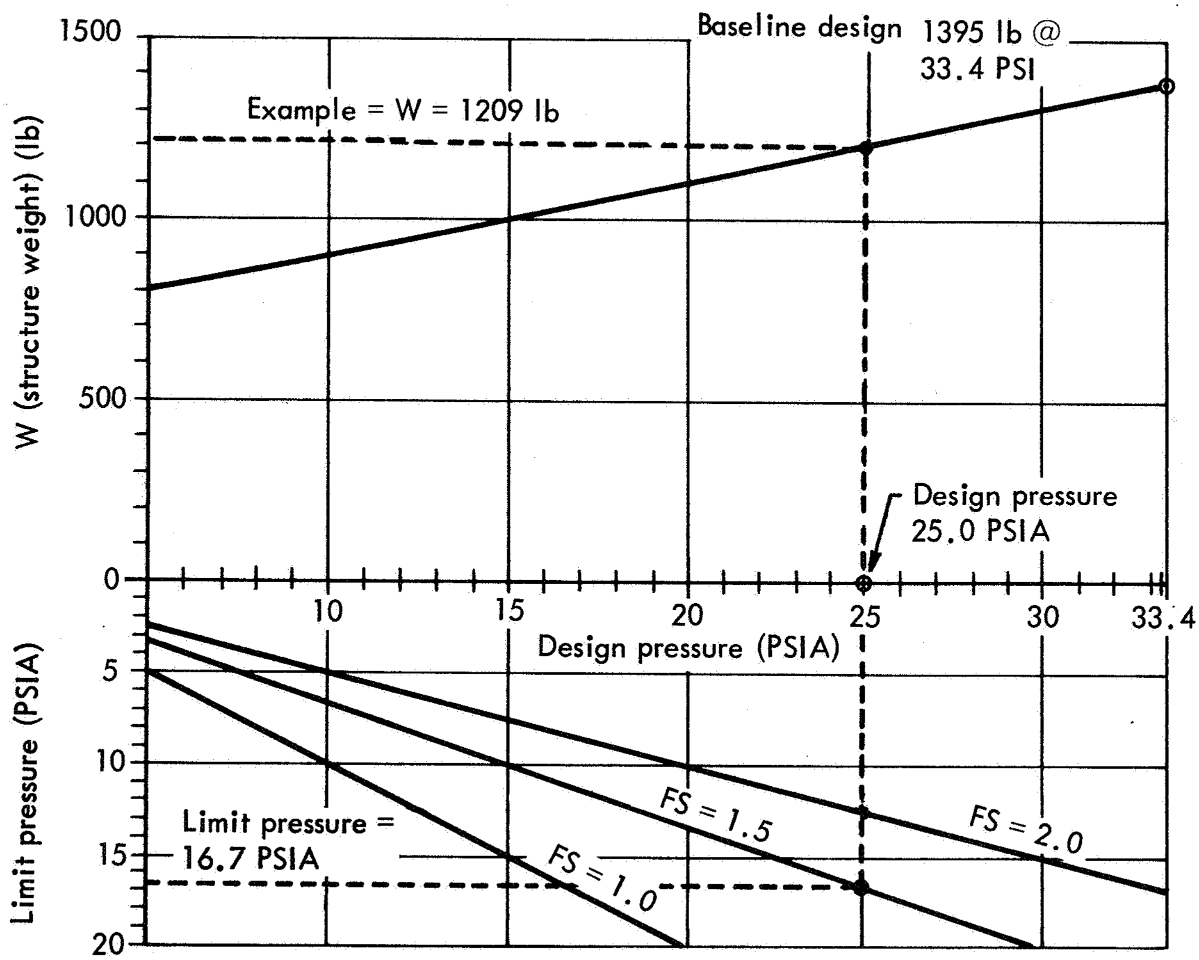


Figure 36. - Effect of pressure and factor of safety on primate structure weight

TABLE 28. - A COMPARISON OF EXAMPLE ADVANCED MATERIALS

Alloy *	Modulus density ratio	Yield strength density ratio
Aluminum Alloys, 2024-T3, 2219-T6 2020-T6	104×10^6 in	500×10^3 in
Magnesium Alloys, HK31A-H24, HM121A-T81	98×10^6 in	445×10^3 in
Magnesium - Lithium Alloy, LA 141	127×10^6 in	400×10^3 in
Lockalloy	380×10^6 in	530×10^3 in
Cross Rolled Beryllium Sheet	650×10^6	894×10^3 in

* Reference 15

Base assembly: The base assembly, illustrated in figure 37, is designed as a single spacecraft subassembly, and provides a convenient focal point for the installation and checkout of nearly all spacecraft subsystems with the exception of the attitude control subsystem which is installed on the pressure shell assembly.

The main structural element of the base is the lower bulkhead assembly, consisting of a single piece of machined aluminum alloy (2219) which is welded to tee section extrusions that form the radial beam system as shown in figure 33. Two machined bosses, located on the z axis just inside the central ring, are the attach points for the two bulkhead tension ties. The bosses are sealed with welded caps after the tension ties are installed. The periphery of the bulkhead is provided with a flanged ring containing the bolt circle for attaching the pressure shell assembly, and with machined fittings which mate with the mounting trusses and with the octagonal frame. Mountings for the cages, feeders, and other items in the pressure vessel are provided by fittings welded to the upper (inside) surface of the bulkhead.

The tension ties, included as part of the lower bulkhead assembly, are installed prior to the installation of the lithium hydroxide can. The ties, fabricated from 17-4PH (H900) stainless steel, terminate in threaded fittings which protrude through bosses in the upper and lower bulkheads. As shown in figure 33, these bosses are closed with welded caps to assure the pressure integrity of the structure.

The octagonal frame assembly consists of a welded 2219-T81 aluminum alloy framework of standard extrusions to which the radial 2024-T4 aluminum alloy sheet metal panels are riveted. The inner edge of the panels terminate at a central sheet metal cylinder which encloses the LiOH can. The assembly is bolted to the bulkhead at the eight peripheral fittings, along the radial bulkhead beams, and at the central mounting ring. Diagonals at panels A, C, F, and H, shown in figure 37, stabilize the structure. The central cylinder incorporates the LiOH can mounting ring.

The base assembly also includes the mounting structure which fastens the spacecraft to the ATM or LMSS racks. The mounting structure consists of four welded tubular truss assemblies, bolted to peripheral fittings on the lower bulkhead and to fittings on the bottom of the octagonal frame and terminate in fittings compatible with the Lunar Module Ascent Stage Separation system. For installations inside the rack, the truss assemblies can be replaced by slippers which attach to the lower bulkhead peripheral fittings and slide on rails mounted to the ATM gimbal ring support structure.

The octagonal frame is enclosed by a series of rectangular flat side panels and trapaziodal bottom panels. These panels are designed as non-load-carrying structures relative to the stability of the frame, and may have a variety of configurations depending on the needs of the equipment mounted on or behind them. Most of these panels are attached to the primary structure through thermal insulators and provide the mounting structure for the multi-layered super insulation. The bottom panels are stiffened by beading, and multi-layered thermal insulation is attached to the inner surface. Side panels

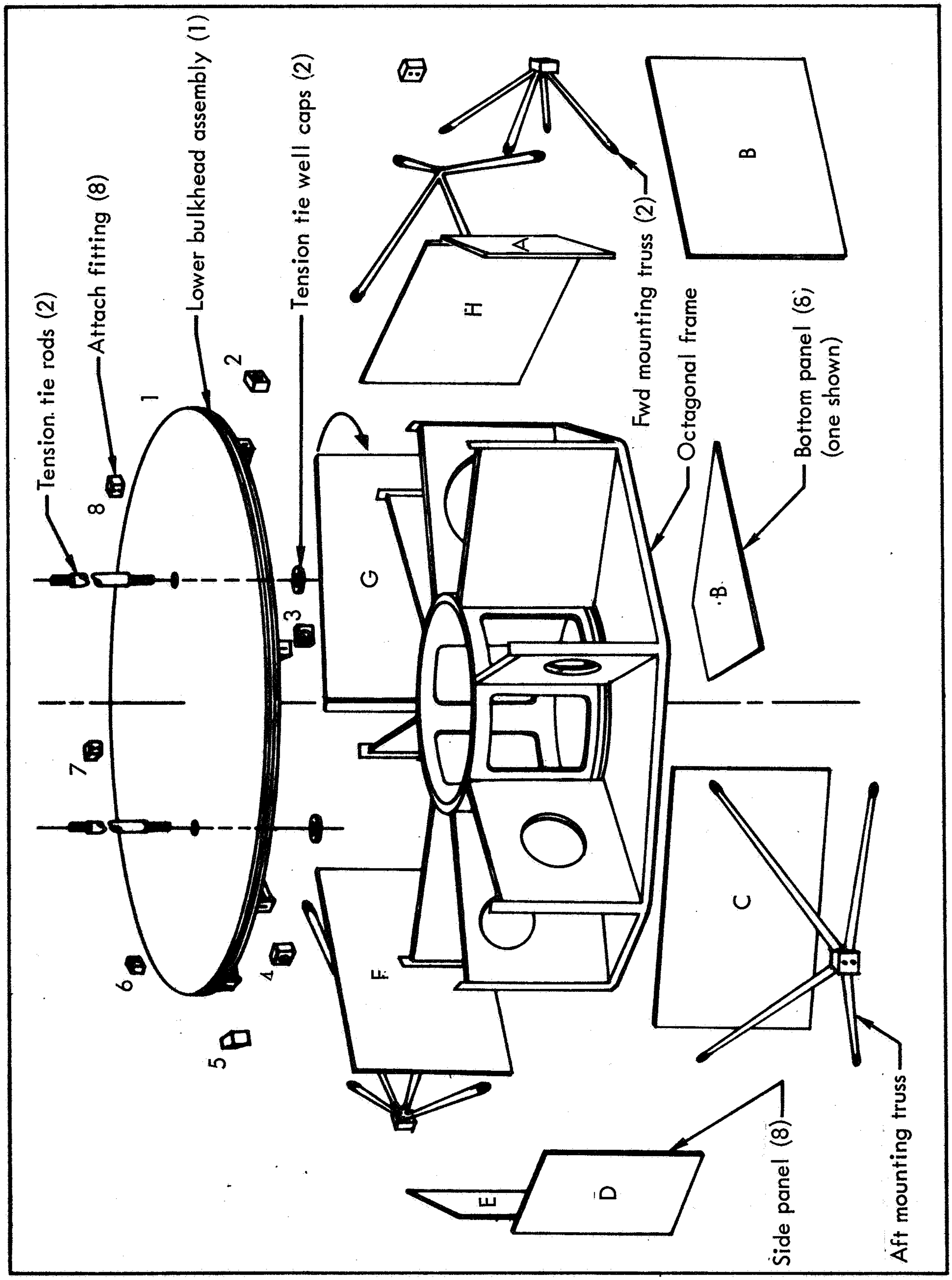


Figure 37. - Base assembly

A, B, G, and H, are heat rejection surfaces for equipment mounted on the inner surface; all are readily removable except for panel G which is hinged. Panel D contains an access door for the fill and drain valves and other items of cryogenic plumbing. These panels will most likely incorporate extruded or formed angles to both stiffen the panel area and serve as mounting surfaces for the equipment. The estimated weight of these panels was based on 0.030 aluminum sheet with the additional weight increments for mounting the equipment and for heat rejection or control, being a part of the equipment mounting budget. Panels C, E, and F are beaded sheet metal with super-insulation on the inner surface.

Pressure shell assembly: The pressure shell assembly, figure 38, is the flat-topped cylindrical structure which comprises the sides and top of the pressure vessel, and on which is installed the Attitude Control system, the Docking Drogue, and parts of the Telecommunications subsystem.

The upper bulkhead assembly is similar to the lower bulkhead except for the non-symmetry in the radial beam pattern due to the location, size and shape of the recovery capsule openings. The center of the bulkhead contains a cylindrical ring to which the docking collar is attached. The tension-tie bosses, located just inside this ring are provided with welded caps to seal the tension-tie penetrations after final assembly.

The pressure shell cylinder, fabricated from welded 2219 aluminum alloy, consists of a machined skin welded to vertical tee section extrusions, with a ring around the bottom for the base-to-shell-attachment bolt circle. Reinforced cutouts in the side of the cylinder accommodate 20 by 20 inch access doors.

The upper bulkhead and cylinder are welded together for a reliable, ultra-low-leakage seal. Analytically, this joint and the one at the bottom of the cylinder are probably the most critical areas in the pressurized structure due to the degree of fixity assigned to the head beams. As a trapaziodally loaded beam running from the edge of the docking collar to the outer edge of the cylinder with both ends fixed, the beam induces very large moments into the cylindrical walls. As a simply supported beam, deflections tend to increase, thereby decreasing the confidence in the pressure seal of the bulkhead/cylinder weld. To resolve this problem the beams were designed as simply supported elements, with small allowable deflections, and with the ends welded into the structure as a semi-fixed end configuration as shown in figure 33.

The docking collar is a machined cylindrical structure, configured at the command module interface, to the same design as that used for the Lunar Module; the details of the design, shown on figure 33, are taken from Apollo Interface Control Document MH01-05128-116, (ref. 16). The docking ring is welded to the upper bulkhead; however, it could be attached to an adapter section by a pyrotechnically operated V-Band clamp to provide a means for separating the spacecraft and command module in an emergency. Such as arrangement would permit changing the docking structure to accommodate different docking systems.

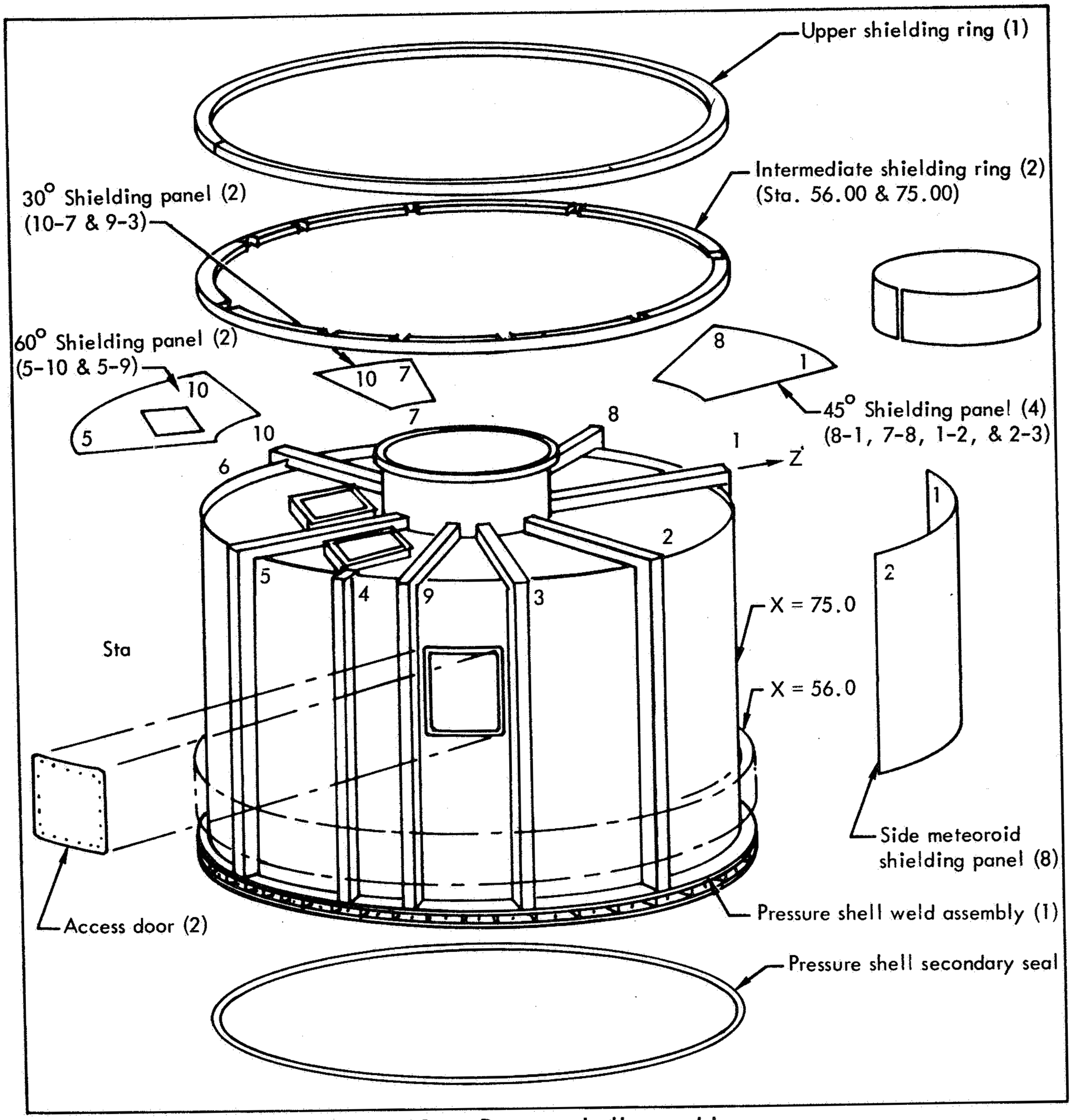


Figure 38. - Pressure shell assembly

The recovery capsule interface is a machined surface located in the recovery capsule wells on the upper bulkhead. Details of the capsule installation are discussed under Life Support Systems.

Two access door assemblies are located in the side wall of the cylinder opposite the loading access doors in the primate cages. These doors are stressed skin elements, their closing or opening requiring the installation or removal of about 40 bolts each and an inspection of the seal integrity; therefore, their final installation is accomplished in the latter phase of the prelaunch operations.

The external upper and side panels provide the necessary meteroid shielding and a support for the thermal insulation. Details are shown in figure 33.

The Environmental Control System radiator is installed between station 44.00 and 55.00 in lieu of the meteroid shielding, details of this installation being shown in figure 33 and discussed under Thermal Control Subsystem.

One of the final operations in the assembly of the spacecraft is the installation and sealing of the pressure shell. The low leakage rates required make the 28 foot peripheral seal between the pressure shell and base of pivotal importance to the success of the mission. Three requirements must be met. First, the seal must be such that the pressure shell can be assembled and disassembled without completing or damaging the seal to facilitate shipping, interface and combined systems checks, and the removal and replacement of equipment items. Secondly, the seal must be designed so it can be completed and then opened and resealed, several times if necessary, without degrading its integrity. This permits incorporating major changes such as the replacement of internal equipment after the final closure is made. The third requirment is that the seal must be designed for a zero-leakage without compromising such structural features as the hard anodize finish on the interior of the pressure shell. The only seal judged acceptable of meeting these criteria with a high degree of confidence was a fusion weld. The fusion weld seal design, discussed in Subsystems Trade Studies (ref. 6) is shown in figures 33 and 39.

The pressure shell to base seal is completed in the following manner. After assuring the seal surfaces are clean and free from debris, the pressure shell and base assembly are aligned and brought into contact at the sealing surface with the bolt patterns aligned; ten bolts, NAS 1351C5-20P or equivalent, are installed, one each in the vicinity of the vertical columns. The remaining bolts are then installed and all bolts torqued to complete the main load carrying portion of the joint. The elastomeric seal in the pressure shell is a secondary seal only. The sealing plate, a .04 thick aluminum band is then fusion welded to the seal flanges as shown in figure 33. The seal is inspected for leakage by pressurizing the void behind the band with a tracer gas admitted through the weld vent holes, these holes being welded shut after the seal inspection is completed. The seal can be opened by cutting the band circumferencially about a half-inch from each flange to expose the bolts. Resealing is accomplished by welding a new band to the edge strips of the previous band and inspecting the weld as before.

As the pressure shell is lowered to the base, the threaded ends of the tension ties are guided through the holes in tension tie bosses which are located in the upper bulkhead. After the peripheral bolt circle has been completed, the 7/8-14 castellated nuts are installed and safetied. The welded caps are then installed and inspected for pressure tightness in a manner similar

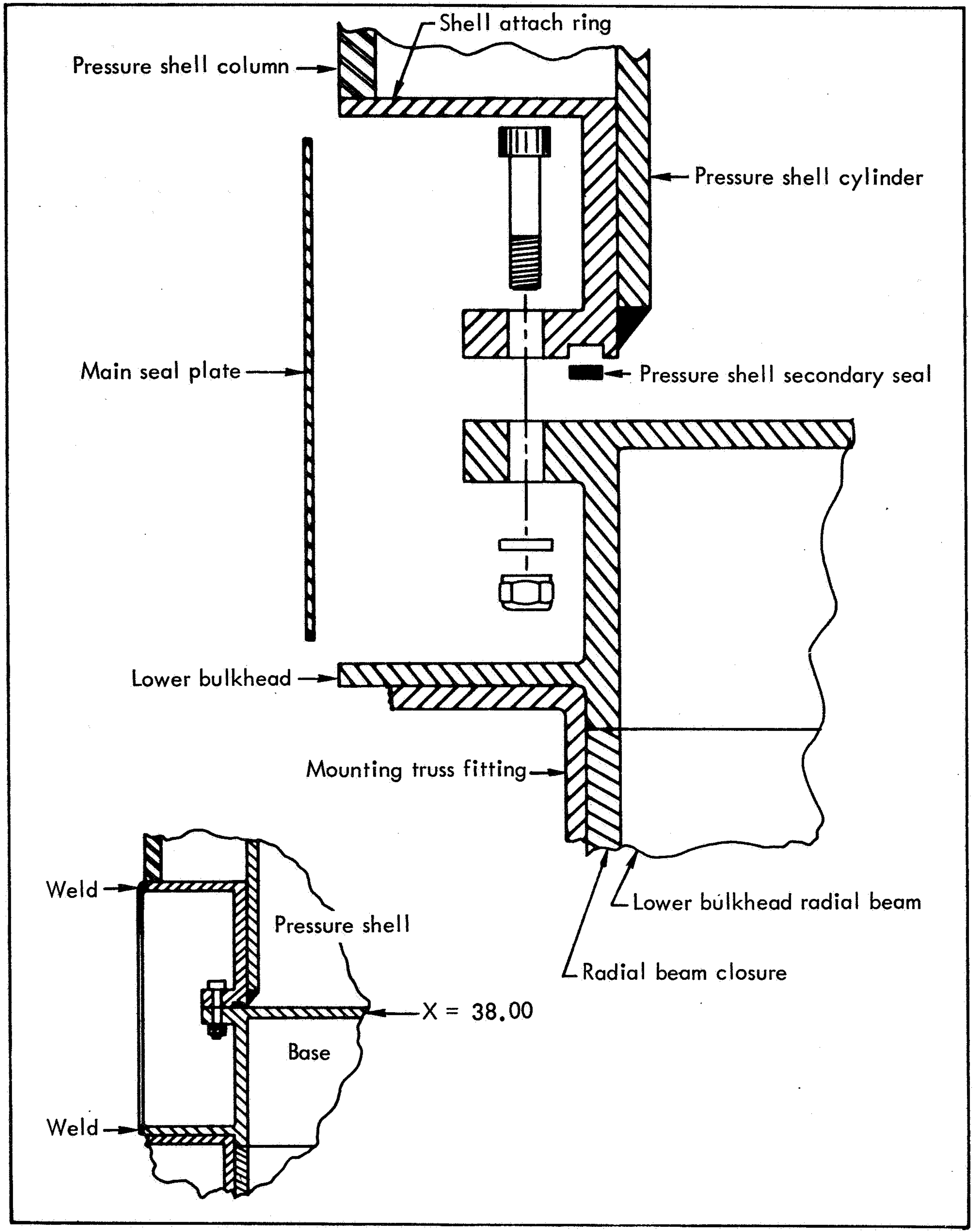


Figure 39. - Pressure shell main seal

to that used for the main base seal. Removal of the pressure shell requires cutting away the caps to expose the nuts.

The pressure-shell-to-base peripheral joint is then completed by first connecting the radiator plumbing and the electrical and electronic cabling and then installing the shielding panels over the seal area as shown in figure 33.

Separation and deployment: The separation and deployment mechanism, figure 40, is identical with the Apollo Lunar Module Ascent Stage (ref. 18), and consists of four explosive nuts and four explosive bolts. If either the nuts or the bolts are fired, separation occurs. The spacecraft is then removed from the vicinity of the Launch Vehicle by firing the SM-RCS to provide about 6 ft/sec delta-V (ref. 17).

Docking mechanism: The docking mechanism, figure 41, is identical with that used on the Apollo Lunar Module. It is installed in the docking collar which is configured to the LM specification as previously noted. The docking mechanism includes the three drogue support fittings which are installed in the docking collar as shown in the Apollo Interface Control Document MH01-05128-116, (ref. 16). The drogue (ref. 19) is attached to the three support fittings and serves as a thermal shield for the docking collar interior after departure of the command module. The drogue assembly, figure 42, consists of a conical structure with the mounting provisions necessary to effect a structural installation in the docking collar. Axial location of the drogue is such that adequate clearance is provided for operation of the handles on the final docking latches from the Command Module. Drogue configuration, envelope dimensions, mounting location, etc., are defined by ICD MH01-05127-116 (ref. 16). The drogue may be removed and stowed in the command module.

Twelve manually operable structural latches are equally spaced about the inner periphery of the command module docking ring to provide a means (when secured) for effecting structural continuity and pressurization capability between the CSM and the docked Primate Spacecraft. Latch configuration, location, installation and interface dimensional criteria are defined by ICD MH01-05128-116 (ref. 20). These twelve latches are circumferentially oriented within the command module tunnel so as not to interfere with probe operation or its support structure. Four of these latches are semi-automatic in function and are designated as "initial sealing" pressure latches. The remaining eight latches are completely manual in operation to complete closure. The initial sealing latches serve to hold the CSM and LEM together with the tunnel pressurized until manual latching (of all twelve latches) is completed to effect a structural bond between the vehicles.

Antenna deployment: The Antenna Deployment Mechanism consists of a damped spring loaded hinge at the antenna base, a pyrotechnic pin-puller, and a series of three latches spaced equidistantly along the antenna mast between the hinge point and the top of the mast (where one of the three latches is located). Antenna deployment is initiated by the pin-puller which releases a power spring that unlatches the mast by withdrawing latch pins from the mast

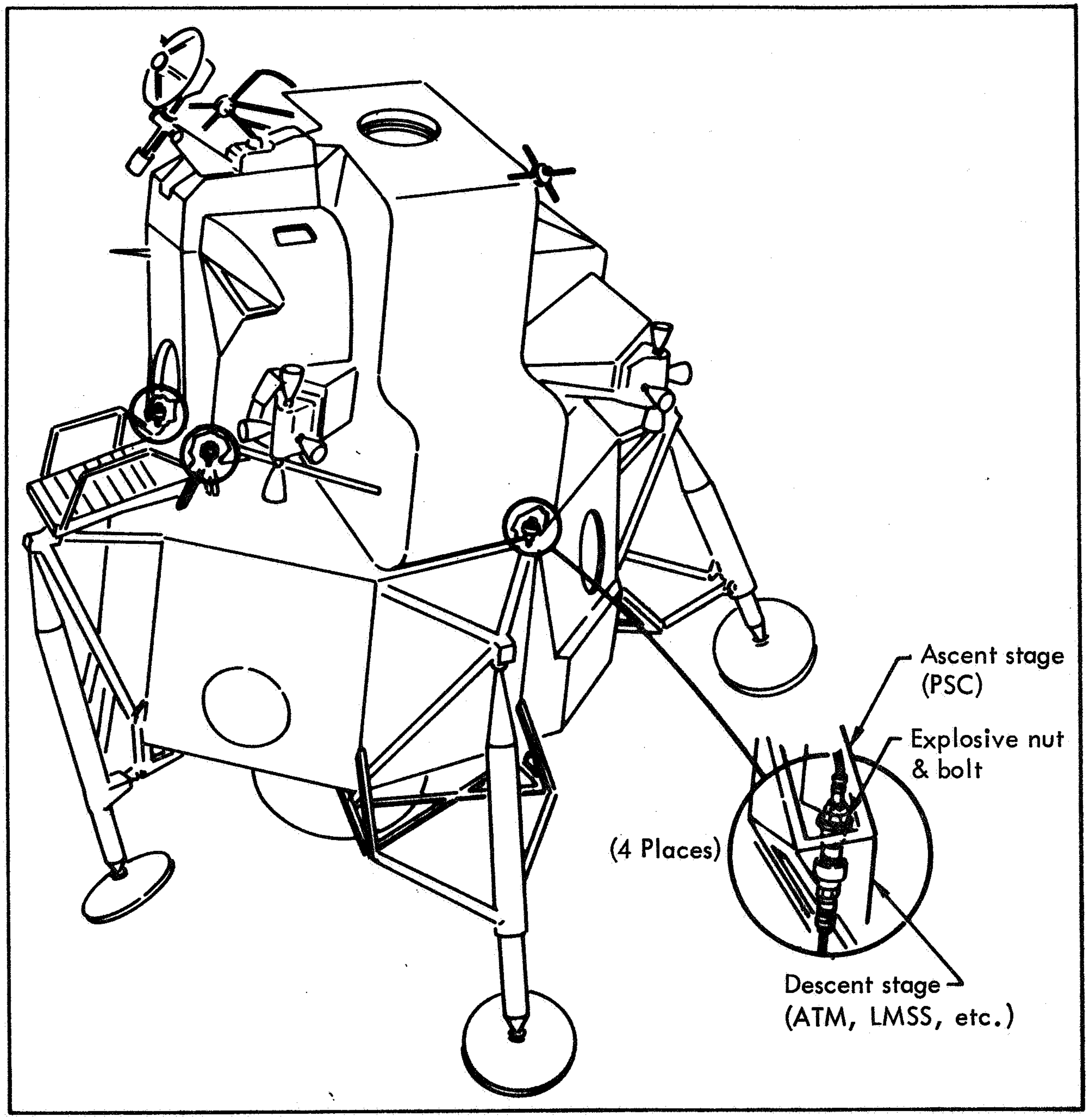


Figure 40. - Separation and deployment mechanism

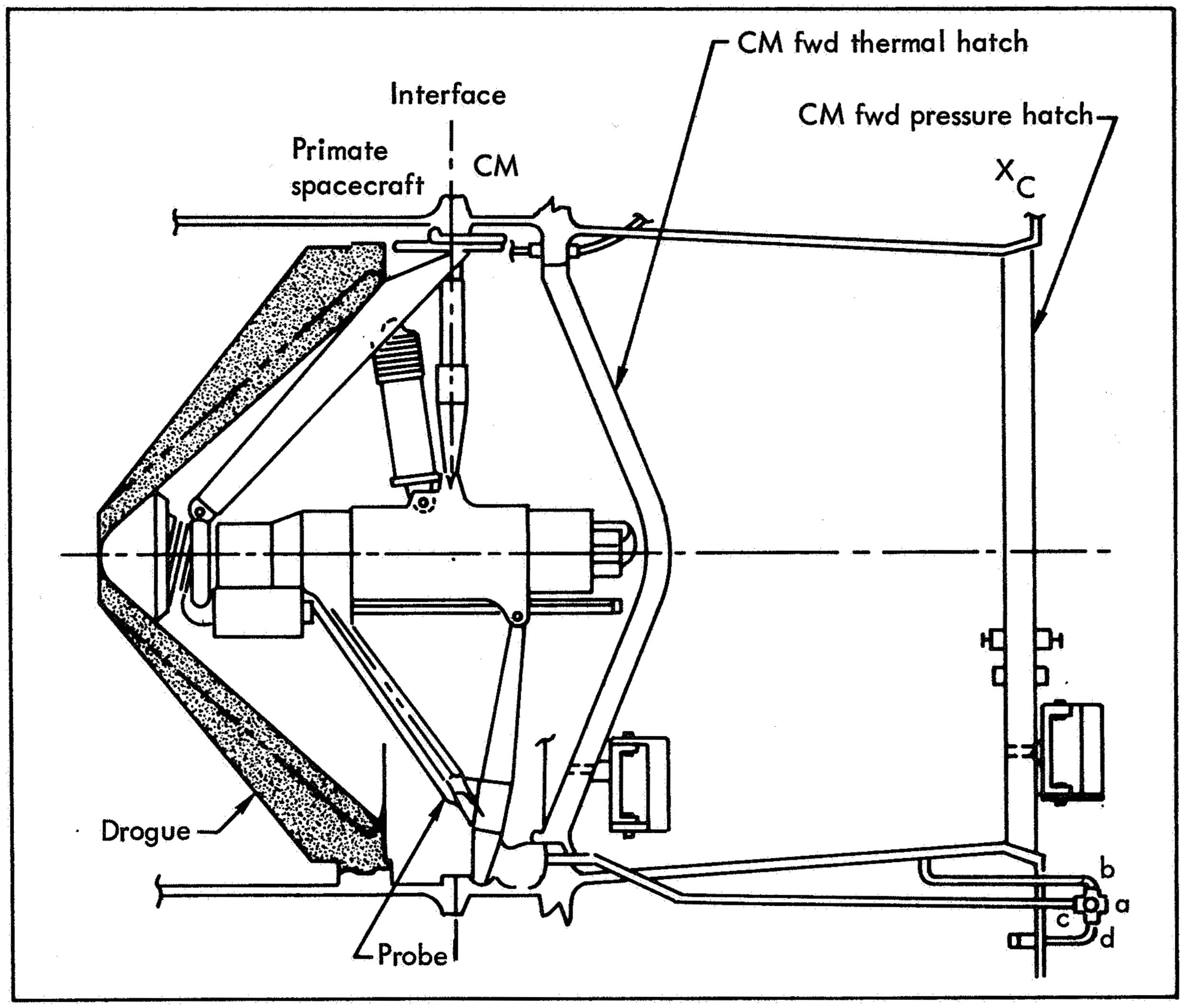


Figure 41. - Docking mechanism

latches, and then unlocking the spring loaded and damped antenna hinge. The mast latches are interconnected by a piano wire type rod that passes through dry lubricated aluminum bushings attached to the spacecraft structure.

Solar panel deployment system: The Solar Panel Deployment system, figure 43, is similar to that used on the Mariner Spacecraft with the exception of the mechanically sequenced latches which restrict deployment to two panels at a time.

Since the size and weight of the Primate Spacecraft panels are about double that of the Mariner, the quantity of the mechanical elements, per panel, has also been doubled to facilitate using existing hardware. In this application, therefore, each of the four solar panels is equipped with two solar panel deployment mechanisms and two solar panel cruise dampers.

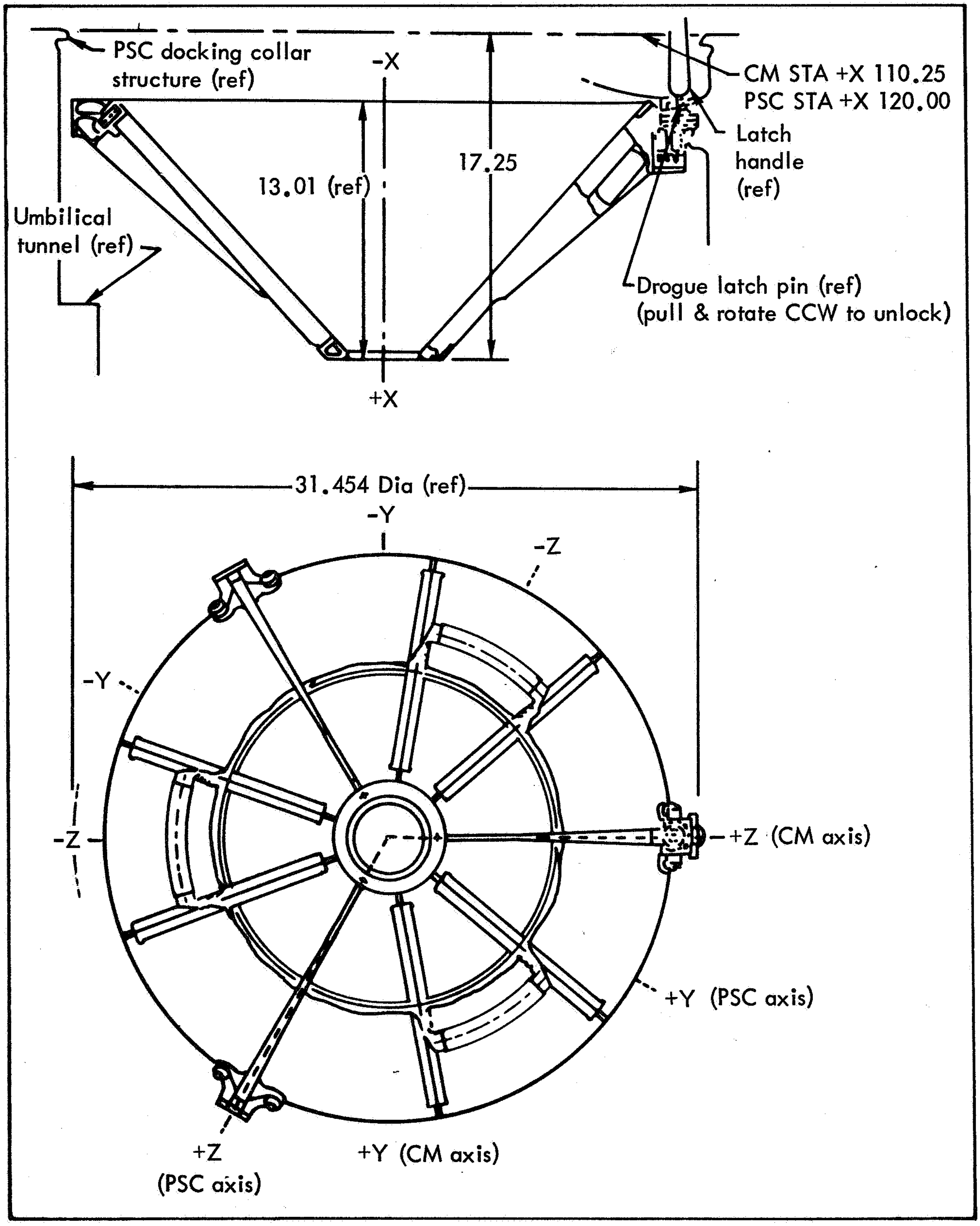
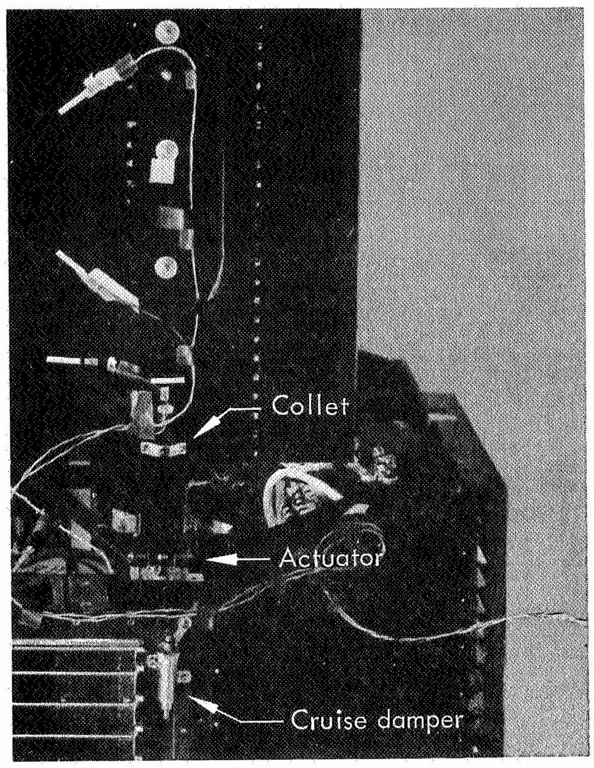
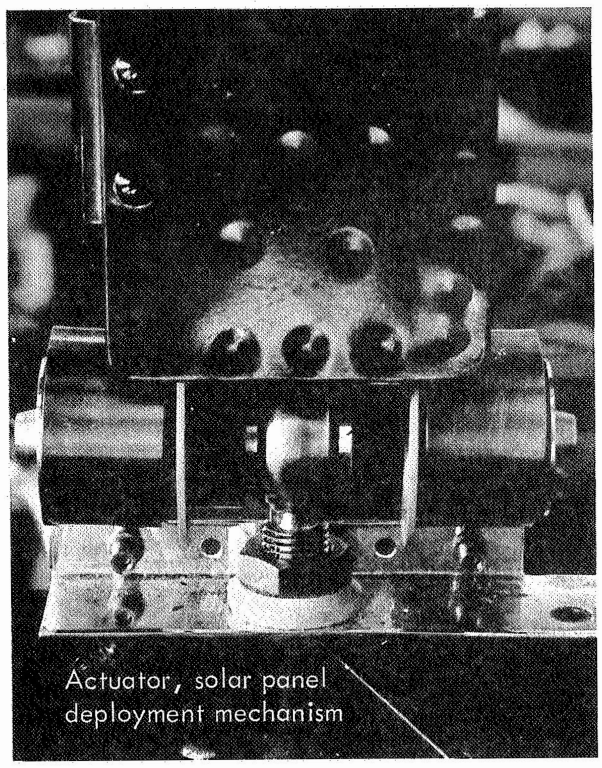


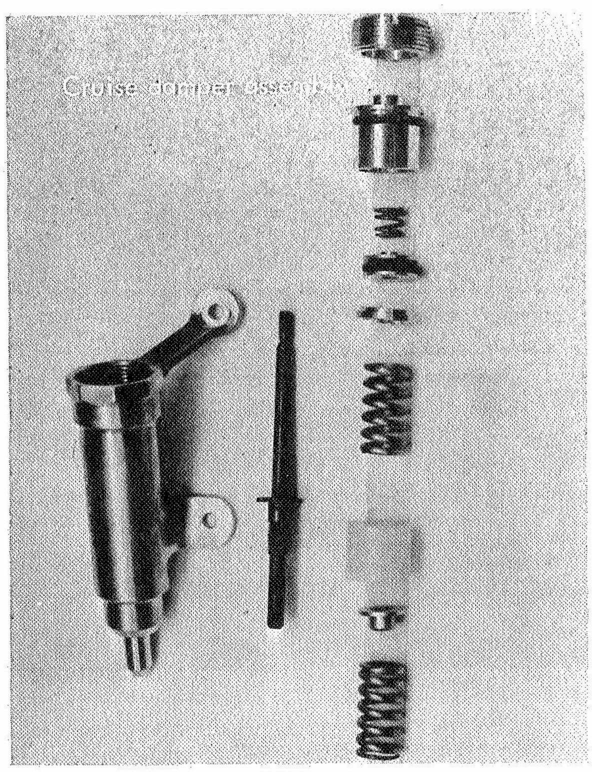
Figure 42. - Drogue assembly



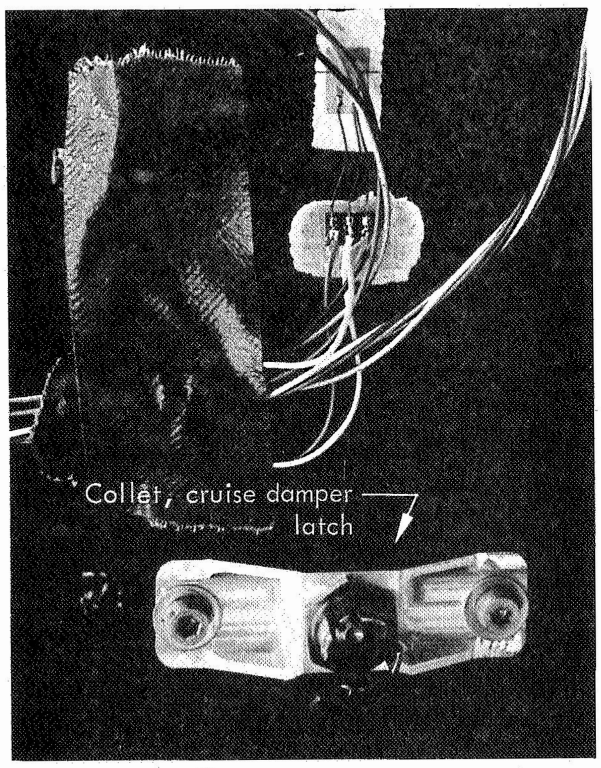
M'69 solar panel deployment mechanism



M'69 actuator



M'69 cruise damper



M'69 collet, cruise damper latch

Figure 43. - Solar panel deployment mechanism

The solar panel deployment mechanism which extends the solar panels is mounted at each end of the solar panel hinge line, each mechanism consisting of dual preloaded clock springs. The mechanisms are identical with those to be used on the Mariner '69 spacecraft (ref. 21) except that the solar panel open event switches and associated bracketry are not installed and the mechanisms are positioned to permit 180 degrees of solar panel travel. The deployment mechanisms are released to extend the panels by a pyrotechnic pin puller. Firing the pin puller releases two solar panel latching rods, one in each of the two outer panels. Each rod passes through three solar panel retention fittings installed on a line parallel to the panel hinge line and passing through the panel's center of gravity. The rods for the outer panels are mechanically coupled together to assure that both panels are released at the same time. As the outer pair of panels reach the fully extended position, they initiate the release of the inner pair by mechanically releasing the inner panel latching rods, this release being accomplished by two mechanical spring-loaded pin pullers, one operated by each of the outer panels and the operation of both being required to release the inner panel latches. Elastromeric dampers installed between the inner pair and the spacecraft structure, and between the inner and outer pair of panels reduce the effects of the launch dynamic environment.

The solar panel cruise dampers arrest the motion of solar panel deployment, and position and damp the motion of the deployed panel. The dampers consist of mechanical-hydraulic dampers located on the outside of the octagonal structure near the solar panel hinges. A self-aligning probe with a low entry force, a high retracting force, and no backlash is provided on each damper. A collet, consisting of an axially split cylindrical tube with internal pawls, is attached to the solar panel structure, and as the panel reaches its extended position, the damper probes enter the collets to latch the panels open.

Advance deployment areas: Generally, the approaches and mechanization of the structure and mechanical subsystem lend themselves to normal engineering development using state of the art materials and techniques.

Preliminary equipment list. - Table 29 presents a list of the components and subassemblies of the Structure and Mechanical Subsystem.

TABLE 29. - STRUCTURE AND MECHANICAL SUBSYSTEM
PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	Structure & mechanical subsystem	NSL	148-13000	1
2	Base assembly spacecraft structure	NSL	148-13100	1

TABLE 29. - Continued

Item	Description	suggested manufacturer	Part no.	Quantity per spacecraft
3	Lower bulkhead assembly	NSL		1
4	Octagonal frame assembly	NSL		1
5	Umbilical support bracket	NSL		1
6	Upper column 1 mounting truss fitting, forward truss	NSL		1
7	Lower column 1 mounting truss fitting, forward truss	NSL		1
8	Upper column 2 mounting truss fitting, forward truss	NSL		1
9	Lower column 2 mounting truss fitting, forward truss	NSL		1
10	Upper column 3 mounting truss fitting, aft truss	NSL		1
11	Lower column 3 mounting truss fitting, aft truss	NSL		1
12	Upper column 4 mounting truss fitting, aft truss	NSL		1
13	Lower column 4 mounting truss fitting, aft truss	NSL		1
14	Upper column 6 mounting truss fitting, aft truss	NSL		1

TABLE 29. - Continued

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
15	Lower column 6 mounting truss fitting, aft truss	NSL		1
16	Upper column 7 mounting truss fitting, aft truss	NSL		1
17	Lower column 7 mounting truss fitting, aft truss	NSL		1
18	Upper column 8 mounting truss fitting, forward truss	NSL		1
19	Lower column 8 mounting truss fitting, forward truss	NSL		1
20	Forward mounting assembly panel A	NSL		1
21	Forward mounting assembly panel B	NSL		1
22	Aft mounting truss panel C	NSL		1
23	Aft mounting truss panel F	NSL		1
24	Side panel A	NSL		1
25	Side panel B	NSL		1
26	Side panel C	NSL		1
27	Side panel D	NSL		1
28	Side panel E	NSL		1

TABLE 29. - Continued

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
29	Side panel F	NSL		1
30	Side panel G	NSL		1
31	Side panel H	NSL		1
32	Bottom panel A	NSL		1
33	Bottom panel B	NSL		1
34	Bottom panel C	NSL		1
35	Bottom panel D	NSL		1
36	Bottom panel E	NSL		1
37	Bottom panel F	NSL		1
38	Bottom panel G	NSL		1
39	Bottom panel H	NSL		1
40	Solar panel mounting bracket (inner)	NSL		4
41	Solar panel mounting bracket (outer)	NSL		4
42	Tension tie	NSL		2
43	Tension tie well cap	NSL		2
44	Pressure shell assembly	NSL		1
45	Pressure shell weld assembly	NSL		1
46	Left hand access door	NSL		1

TABLE 29. - Continued

Item	Description	Suggested manufacturer	Part no.	Quantity per spacecraft
47	Right hand access door	NSL		1
48	Upper meteoroid shielding ring	NSL		1
49	45° shielding panel	NSL		4
50	30° shielding panel	NSL		2
51	Left hand 60° shielding panel	NSL		1
52	Right hand 60° shielding panel	NSL		1
53	Intermediate meteoroid shielding ring	NSL		1
54	Side meteoroid shield ing panel	NSL		8
55	Secondary pressure shell seal	NSL		1
56	Pressure shell main seal plate	NSL		1
57	Panel A seal shield	NSL		1
58	Panel B seal shield	NSL		1
59	Panel C seal shield	NSL		1
60	Panel D seal shield	NSL		1
61	Panel E seal shield	NSL		1
62	Panel F seal shield	NSL		1

TABLE 29. - Concluded

Item	Description	Suggested manufacturer	Part no.	Quantity per spacecraft
63	Panel G seal shield	NSL		1
64	Panel H seal shield	NSL		1
65	Tension tie well cap	NSL		2
66	Separation and deployment system	See Grumman Aircraft for details, same part as on LEM ascent stage		1
67	Docking drogue	See North American Aviation for details, same as on LEM ascent stage	V28-575201	
68	Solar panel deployment mechanism	NSL Actuators, pin puller, cruise dampers & latch assemblies same as Mariner '69 - may be pruchased		
70	Antenna deployment	NSL		2

Instrumentation

The primary function of the instrumentation system is to acquire the data from the primate experiments and spacecraft supporting subsystems and to condition and convert such data into a format acceptable by the telemetry and RF subsystems for subsequent transmission to earth.

The acquisition function is accomplished by a variety of transducers and sensors varying from complex gas mass-spectrometers, television cameras, biotelemetry systems, to simple temperature measuring thermistors and status sensing microswitches.

The signal conditioning function is performed by a centralized assembly which varies in sophistication from bi-phase demodulation of gryo microsyn signals to simple passive voltage divider attenuators used in the electrical power subsystem. All conditioned signals are presented to the telemetry subsystem as 0 to 5 volts dc except digital derived data which is processed through pulse-shaping circuitry in the data processor before being interlaced with the PCM data stream.

The instrumentation section, therefore, defines the requirements of instrumentation, preceding the telemetry and RF subsystems. A functional description of each subsystem component is given and the various development areas requiring further study and definition are discussed.

The subsystem from which the data are acquired are as follows:

(1) Life support subsystem
(2) Thermal control subsystem
(3) Structure and mechanical subsystem
(4) Instrumentation subsystem
(5) Telemetry subsystem
(6) Command and control subsystem
(7) Electrical power subsystem
(8) Attitude control subsystem.

Instrumentation requirements. - The instrumentation requirements are defined in the MEI Specification No. CP-10000 (ref. 7). For clarification, these have been extracted, and are classified under data measurement requirements and instrumentation requirements.

Data measurement requirements: The requirements for data measurements are listed below.

(1) Provide during prelaunch and orbital phases, data gathering and transmitting spacecraft equipment for selected physiological and behavioral data to permit documentation of biological changes due to weightlessness and to permit monitoring of the well being of the animals.

(2) Provide data gathering and transmitting spacecraft equipment that will supply scaling factors for oxygen, water, food, contaminant and temperature

control for future long term life support systems operating in a weightlessness space environment, during launch and orbital phases.

(3) Provide data gathering and transmitting spacecraft equipment that will supply long term life support component experience by monitoring appropriate equipment performance parameters, during launch and orbital phases.

(4) The status of the spacecraft and experiment animals will be provided to the astronauts during the recovery operation. Critical items that may be hazardous to the astronauts or the animals will be directly displayed to the astronauts.

Engineering data will be gathered for assessment of the operation of the environmental control unit both for recognition of incipient failure and for evaluation of system concepts and component reliability for future designs.

These data will include, but not be limited to, measurement of total pressure, oxygen partial pressure, temperature and humidity.

The feeder will signal delivery of a pellet such that the time may be recovered to the year, month, day, hour, minute and second within 10 milliseconds.

Engineering data will be gathered for assessment of the operation of the feeder both for recognition of incipient failure and for evaluation of system concepts and component reliability for future designs.

The waterer will signal delivery of an aliquot such that the time may be recovered to the year, month, day, hour, minute and second within 10 milliseconds.

Engineering data will be gathered for assessment of the operations of the waterer, both for recognition of incipient failure and for evaluation of system concepts and component reliability for future design.

The behavioral panel will signal onset of stimuli, handle actuation and proximity switch actuation such that the time of each may be recovered to the year, month, day, hour, minute and second within 10 milliseconds. Once a day a loud, 110 db, noise will be introduced by ground command for one second duration to produce a startle response which will be monitored on television. Engineering data, aside from that specified above, will be gathered for assessment of the operation of the behavioral panel, both for recognition of incipient failure and for evaluation of system concepts and component reliability for future design.

The mass/volume measurement device will have an accuracy of ±1% of true mass as received at the ground station. Engineering data will be gathered for assessment of the operation of the device both for recognition of incipiend failures and for evaluation of system concepts and component reliability for future design.

The instrumentation subarea will provide the transducers and signal conditioning necessary for the monitoring of the animals and the status of the other subareas. The transducers do not include the implanted animal sensor/transmitter but will include the receiving portion for the animals ECG, temperature, and respiration rate.

For television monitoring of the animals, the lighting system will furnish 25 foot-candles of illumination for 14 hours and 0.1 foot candle of illumination for the remaining 10 hours per day. The photoperiod of 14 hours on and 10 hours off will be changed to 24 hours on and 24 hours off upon ground command.

The television camera will provide sufficient resolutions, frame rate and shades of gray to provide information about the health of the primate and his apparent condition. The television transmission will be compatible with the recorder capabilities and the down link bandwidth and time constraints

Once daily the radiation counter will measure the accumulated irradiation dose. The counter will be capable of measuring in the 0 to 5 rad region with possible excursions of 50 to 1000 rads. The maximum permissible error for data received is 10%.

The activity counter will monitor the motion of the primate in the Life Cell and register frequency of activity.

Temperature and ECG data will be telemetered from the animal to the receivers located in or near the life cell. The animal's heart rate is 257 ± 31 beats per minute with a range of 160 to 333 beats per minute. Temperature is in the range of 95 to 105°F. The electrocardiograph will be monitored either in ten second periods every five minutes, or in five minute periods six times per day. The ECG waveform shall be reproduced with sufficient fidelity to permit (upon receipt at ground station) measurement of the intervals and amplitudes depicted by figure 44 within 0.01 second and 0.1 millivolt respectively. Temperature resolutions shall be ±0.1°C maximum and long term accuracy shall be 1°C maximum.

For the physiological sensors, the maximum error for the mass measurement received at the ground station will not exceed 1 percent.

Furthermore, animal vocalization will be recorded in the frequency range of interest from 20 to 20,000 Hz with 50 to 12,000 Hz acceptable.

Engineering sensors and transducers will be incorporated in all spacecraft subsystems to measure the subsystem status operation and performance. Use of voltage, current, and frequency pickoffs will be provided for in the subsystem electrical design. The noise level in the life cell will be maintained less than 50 db at all frequencies.

Data processor: The data processor will acquire switch closures and impulses from the behavioral panel, the feeder, the waterer, the life cell mechanixm, the programmer, and central timing and will convert such pulses

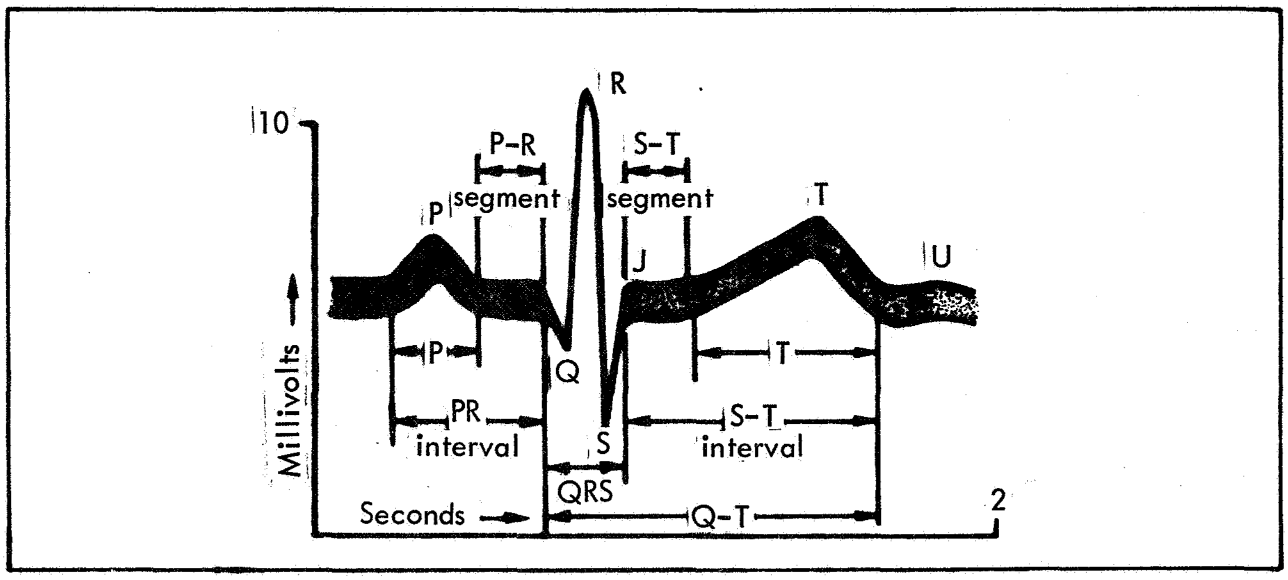


Figure 44. - ECG wave form

into binary words representing start or stop times in time reference code or binary words representing time span of accomplishment.

Such generated binary time words will be stored in buffer registers until readout by the digital multiplexer of the telemetry subsystem. All subsystem status events, realized as switch closures or impulses will be delivered to the data processor wherein each event will be encoded in a pseudobinary word, its identity given by the bit location, the occurrence by change to a 1 bit state. The events binary words will be stored in registers and read out by the digital multiplexer at least once each minute.

Behavioral regimen: The general procedure for the behavioral regimen employs a multiple schedule (MS). In the MS derived for the animals in the spacecraft, different stimuli signal the occasion for different behaviors. This particular MS has six components, four with visual stimuli controlling food or water rewarded responses, one of which has an auditory discriminative stimulus, a component involving avoidance of a noxious stimulus, and an off period. The four food or water reward components--timing, vigilance, interlock, and exercise--alternate with each other and with an occasional off period according to the sequence listed in table 30. The avoidance component overrides this sequence occurring only four times each day, 15 minutes, 5, 9, and 13 hours after the lights come on. If no correct response occurs within eight daylight hours, the schedule moves to the next component. The only limit to amount of food and water consumed by the primates is placed by the amount of work required.

On the behavioral panel, illumination of the blue light around the TIM handle timing component, signals the TIM behavior. The correct behavior is to space responses at least a specified minimum number of seconds apart. The first response after the TIM stimulus light illuminates starts the timing interval. The second response, if it exceeds the time interval, enables the food and water lip switches, illuminates their stimulus lights, and terminates the TIM light. If the second response does not exceed the minimum time, the timing requirement is reset and restarts. A response is defined as an upwards activation of the handle at least 10 milliseconds in duration. The handle must be released for a second response to occur. The primate is not required to hold the handle and probably will not do so. The minimum time interval between responses is adjustable upon ground command to the following five preset points: 5, 15, 30, 45, 60 seconds. The component continues until a correct response is made.

TABLE 30. - SCRAMBLED SEQUENCE FOR THE FOUR COMPONENT MULTIPLE SCHEDULE

TIM, VIG, ILK, OFF, EXC, EXC, ILK, VIG,
TIM, ILK, ILK, TIM, EXC, OFF, VIG, VIG, EXC, OFF,
TIM, TIM, EXC, VIG, ILK, TIM, TIM, ILK
ILK, VIG, EXC, EXC, OFF, ILK, EXC
TIM, VIG, VIG, repeat.

The vigilance component is signalled by illumination of the yellow stimulus light around the VIG handle, and indicates the VIG behavior. The correct behavior is to respond (as defined previously) when the frequency of an intermittent auditory click increases slightly. The reference click rate should be approximately six per second and the test click rate approximately eight per second. The reference remains on for two seconds and off for two seconds. The test click rate also remains on for two seconds. The test rate click rate should occur a variable period of time after the VIG component begins. A satisfactory sequence of variable time periods is: 28, 4, 16, 12, 40, 8, 24, 8, 8, 4, 20, 24, 12, 8, 20, 4, 44, 12, 8, and then repeat. If a response is made during the reference click, it has no effect. The reference click rate returns again after another period averaging 16 seconds. The component continues until a correct response is made which terminates the VIG light, etc.

For the interlock component, a red stimulus light illuminates preceding the interlock task (ILK). The primates correct response is to activate the ILK handle a number of times rapidly or a fewer number of times over a longer period of time. This is to say, the task is interlocked between primate responses and time with at least one primate response required for successful completion of the task. The total number of responses plus time is variable upon ground commands from 1, 10, 35, 50, 75 to 100 seconds plus responses.

Illumination of the green light around the EXC handle signals the exercise component, EXC, behavior. This task consists of alternately pushing up and pulling down an overhead handle across an excursion of approximately 20 inches. This distance should be tailored to the individual primate in order to encourage maximal flexion and extension of trunk, arms, and legs.

To allow a smooth adjustment to weightlessness and to automate acquisition of the response, the level of effort required must be variable by ground control, being reduced to a comparatively very low level upon launch and during training, and then brought up to high levels gradually or incrementally each day or so. Ground control will in this manner allow some degree of control over and adjustment of the amount of exercise performed in flight.

Because of the requirement for resistance to movement in both directions, and the requirement to vary the force necessary, an ergometric, magneto, dynometer type device would probably be most suitable, although it is conceivable that other devices, such as a variable orifice shock absorber would be feasible. A successful response would be defined as x gram-seconds in the former device ground adjustable to four levels, or y pushes and pulls in the latter ground adjustable to 1, 2, 4, 8. The variable resistance should encompass the range from 1/8 to 5 pounds in approximately ten increments roughly proportional to the absolute level, i.e., 1/8, 1/4, 1/2, 3/4, 1, 1.5, 2, 3, 4, 5 pounds.

It would be very desirable to occasionally present some noxious stimulus, avoidance component, AVD, such as wind or electric shock which the primates could avoid or escape. If a weight, mass, volume, density, etc., measurement device can be devised, the desired response would be entrance into this area upon auditory cue preceding onset of the noxious stimulus by ten seconds. If a weighing device proves unfeasible, then a handle response like the TIM, VIG, and ILK would be used. If no noxious stimulus can be provided, but entrance into a weighing device is needed, this can be accomplished by requiring entrance in order to approach or reach the response panel. One wall of the weighing device could be the response panel itself, thereby requiring the primate to be in position for weighing in order to work.

The temporal sequence of stimulus and response events must be preserved and related to the time of day, within one second. Some intervals between events require further time resolution to 0.01 second.

(1) TIM:

(a) The interval between stimulus onset and the first response

(b) Between each response

(c) Response duration

(d) After 100 seconds (a) and (b) can degrade to seconds

(2) VIG:

(a) The interval between a response and the onset of the preceding reference stimulus, if a test stimulus has not intervenced

(b) Auditory test stimulus onset to the next response

(c) Response duration

(d) After 100 seconds (a) and (b) can degrade to seconds

(3) ILK:

(a) The interval between stimulus onset and the first response

(b) Between each response

(c) Response duration

(d) After 100 seconds (a) and (b) can degrade to seconds

(4) EXC:

(a) If the device is an event type mechanism identical to TIM and ILK, a response is a push and/or a pull

(b) If an analog output, analog recording to 1 KHz, or if analog to digital conversion, 8 bit resolution sampled 100 per second beginning with response onset and continuing as long as the output is above threshold.

(c) Stimulus onset to first response

Errors, or inappropriate responses such as an ILK response during TIM, or premature activation of a lip switch should be resolved to seconds.

In addition to adjustments of amount of work required on each task, provision should be made to:

(1) Repeatedly skip any or any combination of components.

(2) Reinstate components.

(3) Sequence advance command.

(4) Deliver reward at completion of response requirement without requiring a lip switch response.

(5) Leave food and water available indicators on, delivering appropriate reward for each lip switch response without requiring a preceding response. In other words, the multiple schedule is not in effect.

(6) Deliver alternately a free food or water reward every five minutes during light on hours.

(7) Deliver no rewards, but otherwise a normal sequence, until each component has occurred, except those being skipped.

(8) Instead of each correct response stepping the multiple schedule on to the next component a command could require 2, 4, or 8 correct responses in each component before stepping. Rewards are still available for each correct response.

Instrumentation description and performance. - The following section will discuss in detail the performance characteristics of the components of the Instrumentation Subsystem and present a specific description of each unit.

However, to give an overview of the entire subsystem, the following general description, along with the Instrumentation Subsystem block diagram, figure 45, is presented.

The Orbiting Primate Spacecraft data being acquired falls into basically two categories -- engineering data supporting the mission and experimental data (the purpose of the mission). Division of the spacecraft subsystems into these two categories follows.

Engineering data:	Experimental data:
Life Support Subsystem	Primate biotelemetry
Thermal Control Subsystem	Television camera
Structure and Mechanical Subsystem	Behavioral panel
Instrumentation Subsystem	Feeder
Telemetry Subsystem	Waterer
Command and Control Subsystem	Life cell
Electrical Power Subsystem	Primate microphone unit
Attitude Control Subsystem	Radiation dosimeter unit
Recovery Capsule Subsystem	Mass/volume measurement unit

The engineering data is acquired by conventional transducers and pickoffs such as pressure transducers, platinum resistor, thermometers, voltage-current pickoffs, and mechanical switches, and the sensor outputs are routed to the central signal conditioning assembly. Here various conversion operations are

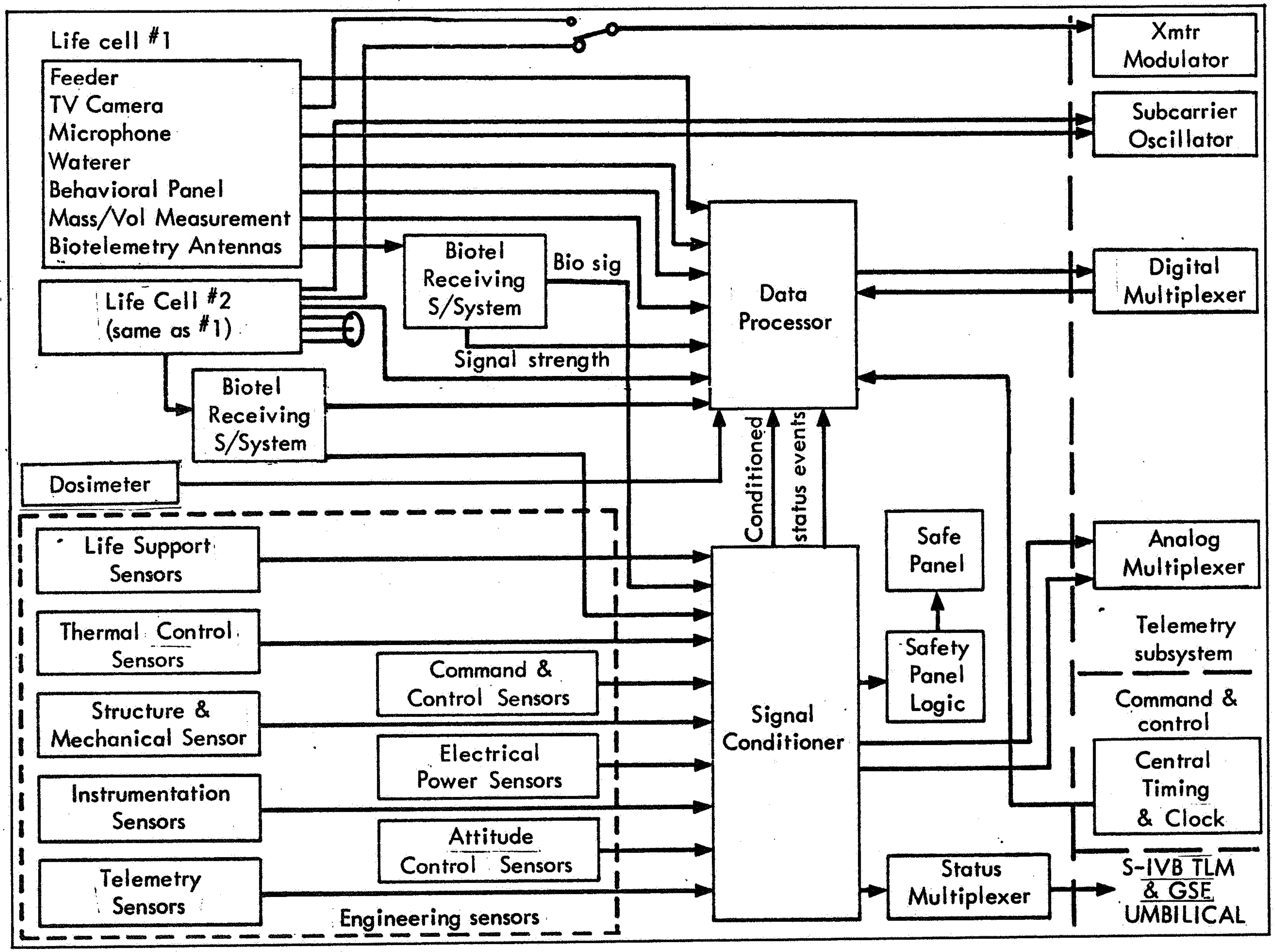


Figure 45. - Instrumentation subsystem

performed on the input signal such as attenuation, amplification, ac to dc conversion, frequency demodulation, and biphase demodulation. The signal conditioner module for each input channel converts the signal into a standard dc high level output voltage of 0 to 5 ±1% volts with a 500 ohm output impedance. These outputs are dual and independent for each channel, and shorting and faulting may occur on either output without affecting the other output. One set of signal conditioner outputs are fed directly to the multiplexers of the telemetry subsystem. The other set of signal conditioner outputs are hardwired to the status multiplexer, the status panel, and the prelaunch checkout umbilical cable.

Some of the experimental data is acquired by conventional transducers and pickoffs and is routed through the signal conditioner to the Telemetry Subsystem as in the case of the engineering data. However, the majority of experimental data bypasses the signal conditioner assembly, since it does not require conditioning, and is routed either to the data processor unit or the data multiplexers of the telemetry system. Thus, switch closure signals from the behavioral panel, the feeder, and the waterer are all routed to the data processor unit where they actuate appropriate binary counters, gates or registers.

Other experimental data, consisting of complex analog waveforms, such as television camera video signal, microphone output signal, and biotelemetry ECG signal are fed directly to the telemetry subcarrier oscillators on the transmitter modulator for frequency modulation.

The dosimeter signal output, 0 - 5 volts dc is fed to the data processor dosimeter accumulation counter where it is converted to the daily radiation dosage as a binary count.

The mass/volume measurement unit signal output is also 0 to 5 volts dc and is routed directly to the analog multiplexer of the telemetry system.

The biotelemetry receiver outputs, after being summed and passed through the filters and discriminators, are fed as separate ECG, body temperature, and respiration rate signals to the analog multiplexers of the telemetry subsystem.

In summary, the Orbiting Primate Spacecraft data, after being secured by the various transmitters and sensors follows three principal routes to the telemetry subsystem.

(1) Either directly to the subcarrier oscillators or transmitter modulator
(2) or through the signal conditioner to telemetry
(3) or through the data processor to telemetry.

The data acquisition function is divided as to type of data required, experiment data, and engineering data, as previously defined.

The following paragraphs list the data points in each of these categories and references them by data number to the Experiment Data Requirement List.

Each category is discussed separately and indicates the type of transducer or sensor required for the data point.

The engineering sensors were defined as those required in all spacecraft subsystems for measurement of subsystem performance, status, operation, or malfunction.

The spacecraft subsystems covered in the engineering sensor survey are as follows:

(1) The Life Support Subsystem
(2) Thermal Control Subsystem
(3) Instrumentation Subsystem
(4) Telemetry Subsystem
(5) Command and Control Subsystem
(6) Attitude Control Subsystem
(7) Electric Power Subsystem
(8) Structure and Mechanical Subsystem.

Since the environmental control subsystem is being supplied by the subcontractor, AiResearch, the engineering sensors of this subsystem are integral parts of the servo control loops and are also being supplied by AiResearch. Telemetry data is being acquired from these sensors using high impedance isolation amplifier pick-offs and only these will be listed for the environmental control system

Similarly, the attitude control sensors are integral functions of the attitude control servo loops and isolation amplifiers will also be used here for obtaining inputs to the Telemetry Subsystem. By using isolation amplifiers, a fail-safe acquisition technique is employed without upsetting the various servo loop response characteristics.

The Data Requirements Summary in Appendix B, Volume V was reviewed and the engineering sensors and transducers were determined for each subsystem. These are summarized in table 31 indicating the various types of transducers and the total number required for each subsystem.

Based on this summary table requirement, over twenty instrumentation manufacturers were contacted for qualified space transducers. Results of this survey are summarized in Appendix D, which presents a comparison matrix of each type of transducer showing the technical parameters as well as cost, power, and size.

Using the transducer comparison matrix and the Data Requirements Summary, the individual data points were relisted and specific transducers were selected from the matrix and so noted in the list.

The final listing therefore, equates the selected transducer to the data point measured and forms the basis for the preliminary engineering sensor summary in costing and performance. This engineering sensor summary is shown in table 32.

TABLE 31. - ENGINEERING SENSORS SUMMARY

Item	Type transducer	Life support subsystem	Thermal control subsystem	Telemetry subsystem	Command & control subsystem	Electric power subsystem	Attitude control subsystem	Structure & mech. subsystem	Total required
1	Pressure, absolute	2							2
2	Pressure, differential								0
3	Temperature probe, plat-R					4			4
4	Temperature surface, plat-foil			6	5	5			16
5	Events switch, mechanical	6						9	15
6	Events switch, voltage			2					2
7	Voltage pickoff					11			11
8	Current pickoff	4				10			14
9	Frequency pickoff								
10	Isolation amplifier	32	4				15		51
11	Diff. amplifier				2				2
12	Relay switch			5					5
13	Cathode-follower pickoff								0
14	ac/dc freq. converter				1				1
15	Gaseous flowmeter	2							2
	Engineering sensors required per subsystem	46	4	13	8	30	15	9	125

TABLE 32. - SELECTED ENGINEERING SENSOR PRELIMINARY SUMMARY

Item	Data No.	Subsystem location	Parameter description	Qty.	Range	% F. S. acc.	Freq. response	Transducer selected	Transducer characteristics	Mfgr.	Part No.
1	D3 to D35	Life support subsystem	ECS servo-control loop sensors as supplied by AiResearch (Ref. Appendix B, Vol. V)	32	Per ref.	Per ref.	Per ref.	High impedance isolation amplifier	1 meg ohm input imp 100 ohm output	NSL	AI-2
2	D43/D50	Life cell	Ram motor current	2	0 to 1 a.	±5%	dc	Current pickoff	.1 to 1 amp	NSL	POC-1
3	D44/D51	Feeders	Capstan motor current	2	0 to 1 a.	±5%	dc	50 millivolt shunt	.1 to 1 amp	NSL	

TABLE 32. - (continued)

Item	Data No.	Subsystem location	Parameter description	Qty.	Range	% F. S. acc.	Freq. response	Transducer selected	Transducer characteristics	Mfgr.	Part No.
4	D45 D52	Life cell feeders	Drum torque motor time	2	---	---	dc	Shaft-clutch switch	Snap action slip release	Micro-switch	2HS1
5	D46 D53	Life cell feeders	Reel takeup motor time	2	---	---	dc	Shaft-clutch switch	Snap action slip release	Micro-switch	2HS1
6	D47 D54	Life cell feeders	Pull off drum motor time	2	---	---	dc	Shaft-clutch switch	Snap action slip release	Micro-switch	2HS1
7	D55 D59	Life cell waterer	Water storage pressure	2	40-60 psia	±5 psia	dc	Strain gage pressure transducer	0-70 psia IP 0-50 mv OP	Stratham	PL749TC-70D
8	D111 to D114	Thermal control subsystem	Thermal control sensors in servo-control loop as part of ECS	4	Per ref.	Per ref.	Per ref.	High impedance isolation amplifier	1 megohm input imp. 100 ohm output	NSL	AI-1
9	D115 to D118	Structure subsystem	Spacecraft release signal	4	---	---	dc	Microswitch at attach points A,B, C, & D	Snap action 7 oz operate	Micro-switch	2HS1
10	D119 D120 D224 D225	Structure subsystem	Solar panel deployed signal	4	Yes-no	---	dc	Microswitch at hinge points A,B, C, & D	Snap action 7 oz. operate	Micro-switch	2HS1
11	D121	Structure	Spacecraft separation signal from CSM	1	Yes-no	---	dc	Microswitch on hinge arm	Snap action 7 oz. operate	Micro-switch	2HS1
12	D137 D138	Telemetry subsystem	PCM data switch positions	2	Yes-no	---	dc	Voltage-trigger switch	Schmitt-trigger set at switch/over	Bourns	3910
13	D139 D140	Telemetry subsystem	TV TWT amplifier base plate temperature	2	+40 to 200°F	±10%	dc	Plat foil resist. gage	-200° to +260°C 500 ohms	Rosemount	118JY
14	D141 D142	Telemetry subsystem	Data transmitter base plate temperature	2	+40 to 260°F	±10%	dc	Plat foil resist. gage	-200° to +260°C 500 ohms	Rosemount	118JY
15	D143 D144	Telemetry subsystem	PCM unit base plate temp.	2	+40 to 120°F	±5°	dc	Plat foil resist. gage	-200° to 260°C 500 ohms	Rosemount	118JY
16	D145	Telemetry subsystem	Data transmitter selection switch	1	1 or 2	---	dc	Extra pole on latching relay	Powered from SCE	Relay Mfgr.	N.A.
17	D146	Telemetry subsystem	TV transmitter selection switch	1	1 or 2	---	dc	Extra pole on latching relay	Powered from SCE	Relay Mfgr.	N.A.
18	D147	Telemetry subsystem	Calibration source input voltage	1	28 ±1 volts dc	±1%	dc	Precision voltage pickoff	Low loss shielded cable	NSL	POV-6

TABLE 32. - (continued)

Item	Data No.	Subsystem location	Parameter description	Qty.	Range	% F. S. acc.	Freq. response	Transducer selected	Transducer characteristics	Mfgr.	Part No.
19	D148	Telemetry subsystem	PCM unit selection switch	1	1 or 2	---	dc	Extra pole on latching relay	Powered from SCE	Relay Mfgr.	N.A.
20	D149	Telemetry subsystem	TV switch position, 1 or 2	1		---	dc	Extra pole on latching relay	Powered from SCE	Relay Mfgr.	N.A.
21	D150 D151	C & C	Command receiver AGC signal	2	±10 mv	±10%	---	Diff. amplifier	dc/ac/dc type 0-5 vdc output	NSL	DA-1
22	D152 D153	C & C	Command receiver base plate temperature	2	+40 to 120°F	±10%	dc	Plat foil resist.gage	-200 to 260°C 500 ohms	Rosemount	118JY
23	D154	C & C	Command decoder base plate temp.	1	+40 to 120°F	±10%	dc	Plat foil resist.gage	-200 to 260°C 500 ohms	Rosemount	118JY
24	D155	C & C	Programmer base plate temp.	1	+40 to 120°F	±10%	dc	Plat foil resist.gage	-200 to 260°C 500 ohms	Rosemount	118JY
25	D156	C & C	Central timing system temp.	1	+40 to 120°F	±10%	dc	Plat foil resist.gage	-200 to 260°C 500 ohms	Rosemount	118JY
26	D192	C & C	Master clock frequency	1	1 meg Hz	±.1%	1 meg	ac/dc converter precision	Freq. demodulate type	NSL	Custom
27	D170 D171 D195 D196	Electrical power subsystem	Secondary battery internal temp.	4	0 to 200°F	±5%	dc	Plat resist probe	-40 to 160°F 0 - 50 mv	Transonics LEM Qual.	T4578-303
28	D197 D198 D199 D200	Electrical power subsystem	Secondary battery output voltage	4	21 to 35 vdc	±5%	dc	Voltage pickoff resist. divider	Passive-diode protected	NSL	POV-1
29	D202	Electrical power subsystem	Prim/sec. battery select switch	1	on/off	---	---	Additional pole on swtg. relay	Independent power +5 volts	Relay Mfgr.	See power subsystem
30	D203 D204	Electrical power subsystem	Primary battery output voltage	2	20 to 35 vdc	±5%	dc	Voltage pick-off resist. divider	Passive-diode protected	NSL	POV-1
31	D166	Electrical power subsystem	Solar panel top surface temp.	1	-150 to 140°F	±5°	dc	Plat. foil resist. gage	-200°C to 260°C 500 ohm	Rosemont	118JY
32	D159 D160 D193 D194 D201	Electrical power subsystem	Solar panel output current	5	0 to 8 a.	±5%	dc	Current pick-off millivolt shunt	Current limiter protected	NSL	POC-4

TABLE 32. - (concluded)

Item	Data No.	Subsystem location	Parameter description	Qty.	Range	% F. S. acc.	Freq. response	Transducer selected	Transducer characteristics	Mfgr.	Part No.
33	D205 D206 D207 D208	Electrical power subsystem	Battery charger reg. output current	4	0 to 10a.max.	±5%	dc	Current pick-off millivolt shunt	Current limiter protected	NSL	POC-2
34	D161	Electrical power subsystem	Main bus unreg. input voltage	1	24 to 32 volts dc	±5%	dc	Voltage pick-off resist. divider	Passive-diode protected	NSL	POV-1
35	D162	Electrical power subsystem	Main bus unreg. input current	1	0 to 37 a. max.	±5%	dc	Current pick-off millivolt shunt	Current pickoff millivolt shunt	NSL	POC-5
36	D209 D210 D211 D215	Electrical power subsystem	Inverter and regulator base plate temperature	4	+40 to 200°F	±5%	dc	Plat. foil resist. cage-base mnt.	-200°C to 260°C 500 ohm	Rosemont	118JY
37	D163	Electrical power subsystem	115V/400 Hz inverter output voltage	1	115±10% ac	±5%	400 Hz	ac voltage pickoff resist. divider	Resistive coupled	NSL	POVA-3
38	D212	Electrical power	1600 Hz, single phase 28v inverter output voltage	1	24-32 vac	±5%	1600 Hz	ac voltage pickoff resistive divider	Resistive coupled	NSL	POVA-4
39	D213	Electrical power subsystem	Essential regulated bus input voltage	1	27-29 volts dc	±1%	dc	Voltage pick-off resist. divider	Passive-diode protected	NSL	POV-5
40	D214	Electrical power subsystem	2400 Hz inverter voltage output	1	35 volts ac	±5%	2400 Hz	ac voltage pickoff resistive divider	Resistive coupled	NSL	POVA-4
41	D172 to D187	Attitude control subsystem	Servo-control sensors in the attitude control loop (ref. Appendix B, Vol. V)	15	Per ref.	Per ref.	Per ref.	High impedance isolation amplifier	1 megohm input 100 ohm output	NSL	AI-1

As stated earlier, Appendix A presents a technical comparison matrix for each transducer requirement against the various manufacturer selected transducers. The selected transducers are noted at the bottom of the matrix columns. Of the various comparison parameters, cost, space qualification, and transducer reliability were the primary selection factors. Technically, the performance characteristics of each type of transducer were approximately equal. They all met or exceeded the space environmental requirements for the mission.

Standard type transducers, such as pressure and temperature sensors, were selected from suppliers of space qualified instrumentation.

Voltage, current, and frequency pickoffs, which must be incorporated as part of the subsystem electronic design for circuit protective reasons, will be supplied by Northrop Systems Laboratories. Final design of these pick-offs must await the conclusion of the parent subsystem design.

The types of transducers to be used for each experiment data signal are called out in table 33 along with the transducer output and excitation power and synchronization requirements.

The transducers will supply the telemetry system directly. Where buffering is required, it is specified. The coupling of mechanical transducers is by shaft coupling (for rotary pots) and expoxy cementing for the strain gages.

TABLE 33.- EXPERIMENT SENSORS SUMMARY

Signal No.	Transducer type	Input	Outputs	Excitation and sync
D36 D37	Bourns model 3917 voltage sensor	-24 to 1000v @ < 0.7 ma	SPSTNO 560min.	20 to 30 v. @ 50 ma max.
D38 D39	Linear single turn potentiometer in torquer motor	rotary motion	0 - 5 volts	5 vdc ±1%
D188 D189	Strain gage bridge and amplifier	mechanical		
D190 D191	Clock register	on-off switch		Sync from master clock
D105 D106 D108 D109	Switch	mechanical	0 or 5 volts	5 volts ±5%

TABLE 33. - Continued

Signal no.	Transducer type	Input	Outputs	Excitation and Sync
D107 D110	Per feasibility test	mechanical or neumatic	0 - 5 volts	28 vdc±1%
D122 D123 D124 D127 D128 D129	Biotel receiver demodulator	biotel transmitter	0 - 5 volts	28 volts
D125 D130	Schmitt trigger	recover signal strength	5 volt pulses	5 volts±1%
D126 D131	Microphone and amplifier w/vox	acoustic	0 - 5 v 50 Hz to 12KHz	28vdc ±1%
D132 D132A D132B	Dosimeter counter temp. (thermistor) voltage (zener)	radiation heat voltage	0/5v pulses voltage voltage	5volts ±1%
D135 D136	Photometer-TV camera	light	0 - 5 volts ±1%	28 volts
D133 D134	Vidicon TV camera	light	2.6 -.6 volts peak-to-peak	24/31vdc
D41 D42 D48 D49	Magnetic Elec. counter Magnetic Elec. counter	dc pulse dc pulse	dc pulse pulse dc pulse pulse	Prog. Seq. Prog. Seq. Prog. Seq. Prog. Seq.
D56	Magnetic	dc	dc pulse	Prog. Seq.
D57	Electronic counter	pulse	pulse	Prog. Seq.
D58	Acoustic	ac	ac	Prog. Seq.
D60	Magnetic	dc	dc pulse	Prog. Seq.
D61	Elec. counter	pulse	pulse	Prog. Seq.
D62	Acoustic	ac	ac	Prog. Seq.
D63	Capacitive	ac	ac	Prog. Seq.
D64	Capacitive	ac	ac	Prog. Seq.

TABLE 33. - (continued)

Signal No.	Transducer type	Input	Output	Excitation and sync
D66	Magnetic	ac	dc pulse	Prog. Seq.
D67	Magnetic	dc	dc pulse	Prog. Seq.
D68	Incandescent lamp	dc	light	Power supply
D69	Incandescent lamp	dc	light	Power supply
D70	Incandescent lamp	dc	light	Power supply
D71	Incandescent lamp	dc	light	Power supply
D72	Electric switch	dc pulse	light	Power supply
D75	Electric switch	dc pulse	light	Power supply
D192A				
D192B				
D79	Electric counter	pulse	pulse	Power supply
D80	Electric counter	pulse	pulse	Power supply
D81	Electric counter	pulse	pulse	Power supply
D83	Magnetic	dc	dc light	Power supply
D84	Magnetic	dc	dc pulse	Power supply
D85	Incandescent lamp	dc	light	Power supply
D86	Incandescent lamp	dc	light	Power supply
D87	Incandescent lamp	dc	light	Power supply
D88	Incandescent lamp	dc pulse	light	Power supply
D89	Electric switch	dc pulse	light	Power supply
D92	Electric switch	pulse	light	Power supply
D93	Electronic counter	pulse	pulse	Power supply
D96	Electronic counter	pulse	pulse	Power supply

TABLE 33. - (concluded)

Signal No.	Transducer type	Input	Output	Excitation and sync
D97	Mechanical switch	dc	pulse	Power supply
D98	Mechanical switch	dc	pulse	Power supply
D99	Mechanical switch	dc	pulse	Power supply
D100	Mechanical switch	dc	pulse	Power supply
D101	Mechanical switch	dc	pulse	Power supply
D102	Mechanical switch	dc	pulse	Power supply
D103	Mechanical switch	dc	pulse	Power supply
D104	Mechanical switch	dc	pulse	Power supply

Signal conditioning: The basic Instrumentation Subsystem requires only six types of signal condition modules, differential amplifiers, bridge amplifiers, ac/dc converters, frequency demodulators, biphase demodulators, and attenuators. However, to provide for future growth potential, the signal conditioner assembly was sized to handle a total of nine types of modules. The additional cost in fabrication is only 5 percent to provide for this capacity. The cost for qualification remains the same.

This growth potential provides for the conditioning of additional types of input signals not included in the basic instrumentation subsystem. The maximum signal capacity of the signal conditioner assembly is itemized in table 34.

TABLE 34. - SIGNAL CONDITIONER ASSEMBLY SIGNAL CAPACITY

Item	Type signal	Max chan	Signal conditioner module	Channels per module	Module max	Capacity nominal
1	+5 to 50 vdc	105 dc	Active attenuator	7	15	5
2	-7 to -50 vdc	105 dc	Inverter-attenuator	7	15	2
3	0 to 150 vac, 350-3600 Hz	30	ac/dc Converter	2	15	2

TABLE 34. - (concluded)

Item	Type signal	Max chan	Signal conditioner module	Channels per module	Module max	Capacity nominal
4	0 to 250 mvdc	20	dc Diff amplifier	2	10	4
5	0 to 7 vdc	10	dc SE amplifier	2	5	3
6	270 to 5700 ohms	20	dc Bridge amplifier	2	10	4
7	+105 to 150 vac 380 to 420 Hz	30	fm FEQ demodulator	2	15	1
8	.5 to 50 vac, 400 to 800 Hz	10	Biphase ac (AM) demodulator	2	5	1
9	+5 to +50 vac	42	Events active attenuator	7	6	3

Since the different types of signal conditioning modules are interchangeable within their assigned sections of the Signal Conditioner assembly, a high degree of flexibility is available in adapting signal conditioning requirements to changing transducer needs. This then provides for major spacecraft subsystem changes in design without redesign of the signal conditioning assembly.

In establishing the initial design criteria, the following factors were considered. Reliability is of prime consideration in the Orbiting Primate Spacecraft program and is obtained by strict thermal and electrical derating of components as well as employing only simple circuits. The predicted reliabilities for the signal conditioning modules range from .99853 to .99894.

Since conventional components are used, the individual modules average 0.6 pound per module with a maximum volume of 900 cubic inches for a 25 module capacity. Two 25 module assemblies will provide a minimum 200 channel capacity. The total weight for both assemblies is 44 pounds.

Due to the long mission flight times, the power consumption of the ECS becomes significant, and was taken into consideration in selection of low power circuitry for each module. The average power per module is 0.6 watt, including conversion losses. The total power required for both assemblies is 30 watts.

The thermal design of the modules provide for the worst case temperature conditions, as relates to the module-cold plate gradient. Thermal fuzz and mechanical contact to the cold plate are used to obtain optimum heat transfer characteristics.

During the flight test checkout of the instrumentation subsystem, it is required that the power drawn from the measurement source shall have no effect on the operation of the source. To accomplish this, a high conditioning-to-source impedance ratio is provided. Also, fail-safe circuitry is provided in the signal conditioning module such that its malfunction will have no effect on the system under test.

To prevent signal grounding interaction each module is designed for common mode rejection, and grounding procedures are implemented to avoid ground loops when dual outputs are used on the signal conditioning modules. Thus, failure or removal of either of the dual output channels will have no effect on the remaining output channel.

For the signal conditioner modules, the unit accuracy is defined as the ratio of the input to output signals, considering hysteresis, zero and gain stability, and repeatability. The design accuracy for all of the modules is $\pm$ 1.0 percent full scale. The input loading error is minimized by using a high input impedance, $\pm$ 500 Kohms, while a low output impedance, less than 500 ohms, is used to reduce output loading error and noise pickup. Since an inflight calibration system is used on the Orbiting Primate Spacecraft, the signal conditioner modules and power supplies are designed for a 1000-hour-long term stability of less than 1 percent.

To achieve adequate temperature stability, the signal conditioner modules incorporate temperature compensation using either positive coefficient resistors or negative feedback circuitry which is temperature controlled.

In order to handle all varieties of change in quantity, type, and range of measurements, the signal conditioner modules are designed to accept a wide variety of signal inputs with minimum module adjustment. Thus, the dc differential amplifier module can be adjusted for full scale inputs from 20 to 250 millivolts and can handle either single ended or double ended inputs by simple pin-to-pin changes. By using standard module configurations and connectors, the units may be interchanged in chassis assigned locations without the necessity of mother board rewiring.

From the above general guidelines, the following criteria were used in designing the signal conditioner:

(1) The equipment modules must be of plug-in type for trouble-shooting, repairs, and configuration changes.

(2) Each signal conditioner output must be capable of being shorted without damage to the module.

(3) Excitation voltages for potentiometers, strain gages and temperature transducer bridges to be supplied from within the equipment.

(4) Under all conditions, including failure modes, the outputs of the signal conditioners must be limited to voltages between -1 and +7 volts dc in order to protect the telemetry inputs. The standard output is 0 to 5 vdc.

(5) The conditioner must be capable of providing rated outputs into a minimum load impedance of 20 kohms.

(6) The output impedance of all conditioners must be less than 500 ohms single-ended.

(7) The input impedance must be greater than 500 kohms on all modules, except the dc attenuators, which are 20 kohms.

(8) Each conditioner is designed to consume a minimum of power from the power source.

Signal conditioner equipment description: The signal conditioner assembly was designed to handle 25 signal modules per each of two assemblies, providing for a maximum of 260 channels, this remaining capacity being held for growth and spares. Each module can handle two signal input channels, except the dc attenuator module, which handles seven input signals.

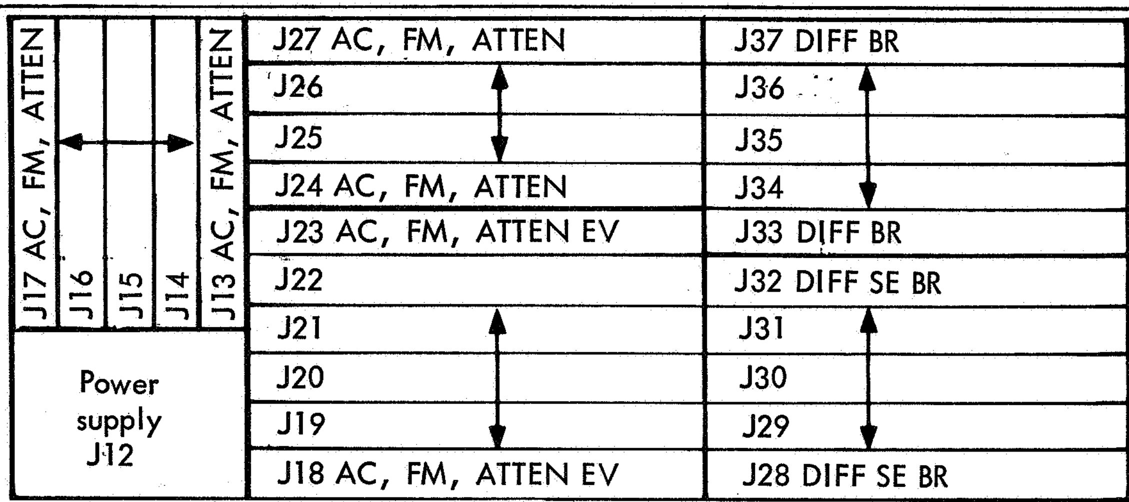
Utilizing the maximum configuration, each signal conditioner assembly can accommodate up to 111 or 130 analog signal inputs depending on the number of dc attenuator modules used. The standard module layout assignment is shown in figure 46. Note that a central power supply is available for converting the 28 volt dc input power to other dc voltages for driving the two dc supplies in J12 which supply power in turn to the modules and the transducers respectively. The assembly input power requirements are 15 watts for the basic design, and 25 watts for maximum capacity.

The functional block diagram of the signal conditioner is shown in figure 47 wherein the various types of signal conditioning modules and their illustrative input sources are shown. A total of nine basic types of modules may be used, such as dc differential amplifiers, dc single-ended amplifiers, dc differential bridge amplifiers, ac to dc converters, low gain dc amplifiers, biphase demodulators, dc active attenuators, frequency demodulators, and dc active attenuator inverters. An illustrative application of these types is shown in table 35.

The signal conditioner assembly unit has a package density of approximately .048 lbs/in^3 and the dimensions are 15.25 by 9.8 by 6 inches. This requires a volume of 900 in^3, not including connectors. The signal conditioner assembly weighs approximately 22 lbs. with each module averaging 0.6 lbs.; the power supply weighs 3.0 lbs.

Television: The television system is composed of two modified Apollo television cameras, each with a two lens turret containing an 80° and a 10° FOV lens. The Apollo television camera is modified by the addition of the two lens turret and the removal of the handle, power switch and automatic light level control switch. The wide angle lens is the Apollo 80° lens while the narrow angle lens is a qualifiable 10° lens.

The sensitivity of the Apollo camera is adequate for the two light levels used in the cage, 25 foot candles and 0.1 foot candle.



AC	AC/DC converter type (-10)
FM	FM demodulator type (-50)
ATTEN	DC active attenuator type (-40)
EV	DC active attenuator event type (-40)
DIFF	Differential amplifier type (-01)
SE	Differential amplifier single-ended type (-02)
BR	Differential amplifier bridge types (-03, -05, -06, -07)

Figure 46. - Signal conditioner module configuration

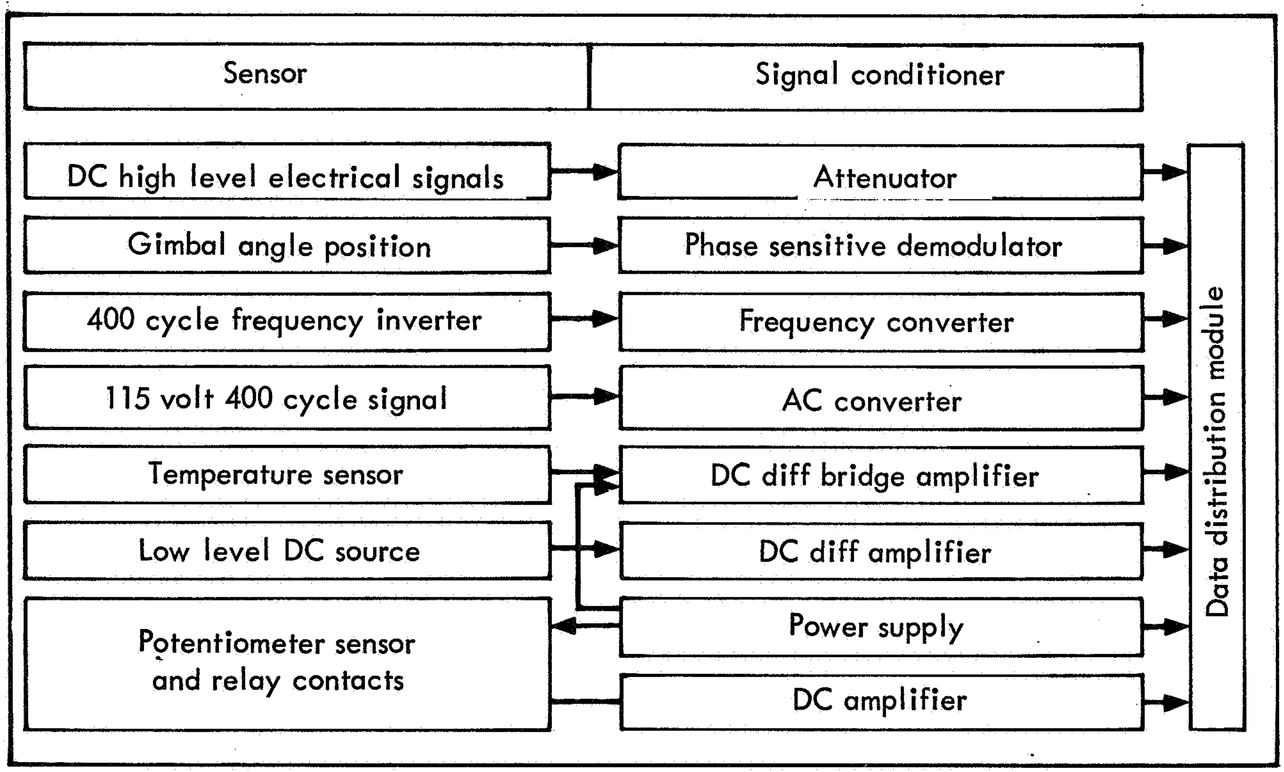


Figure 47. - Signal conditioner functional block diagram

TABLE 35. - SIGNAL CONDITIONER INPUTS

Signal type	Signal characteristics	Conditioner type
Static inverter temperature	32 to 248° F	dc differential bridge amplifier
ac bus voltage	0 to 150 volts	ac to dc converter
Main inverter frequency	380 to 420 Hz	Frequency demodulator
Resolver position	Sine and cosine of pitch-yaw-roll	Biphase demodulator
dc voltage - battery Bus A	0 to 45 volts dc	dc active attenuator
Negative dc supply voltage	0 to -20 volts dc	dc active attenuator inverter
dc current secondary battery	0 to 100 a.	dc differential amplifier

The camera contains a one-inch vidicon type imaging tube capable of resolving 250 television lines on an EIA test pattern with incident vidicon target highlight illumination of 0.1 foot candle.

The equipment operates with a maximum power dissipation of 6.75 watts, with a steady voltage between 24 and 31 volts dc under the following conditions.

(1) Maximum 78 volts, recovery in 10 microseconds, 2 pulse per second repetitive rate: 32 volt recovery in one minute, non-repetitive.

(2) Minimum 21 volt, recovery in one second, non-repetitive.

(3) Ripple, one volt peak-to-peak from 50 to 20,000 Hz.

(4) The dc power shall be independent of equipment ground.

As specified, this performance is not required during abnormal transient or low line voltage conditions. However, equipment remains undamaged as a result of exposure to transient or low line voltage conditions. However, equipment remains undamaged as a result of exposure to transient or low line voltages.

An internal oscillator provides a 320 line frame. The 320 lines shall include those blanked during vertical retrace. The horizontal blanking interval is 10.5 percent maximum of the total line time. During the blanking interval, the waveform begins with a porch of half the amplitude of white to sync tip, level for 4 microseconds minimum. The tolerance of the porch amplitude is plus or minus 15 percent of the white to sync tip level. The waveform then steps to sync level for 14 microseconds minimum and returns to a level midway between white and sync tip for a minimum of 4 microseconds. The rise time is determined by the video filter. The sync tip level is nominally 30 percent of white to reference black.

The internal oscillator provides a frame scan of 10 frames per second. The vertical blanking interval is 2.75 percent maximum of the total frame period (nominally 8 lines). The waveform, during the frame blanking interval, consists of a porch midway between the white and sync tip for a minimum of 4 microseconds at the beginning of 4th frame blanking interval, stepping to sync tip level for 2,500 microseconds nominally and returning to a porch level midway between white and sync tip level for a minimum of 4 microseconds. The tolerance of the porch amplitude is plus or minus 15 percent of the white to sync tip level. The rise time is determined by the video filter. The wave form is illustrated in figure 48.

The video response characteristics of the equipment is within plus or minus 3 db of the video output specified from 40 Hz to 400 KHz, and within plus 3 or minus 12 db from 400 KHz to 500 Hz. The roll-off above 500 KHz is not less than 20 db per octave.

The equipment is capable of rendering a minimum of 5 gray scales under the illumination levels specified.

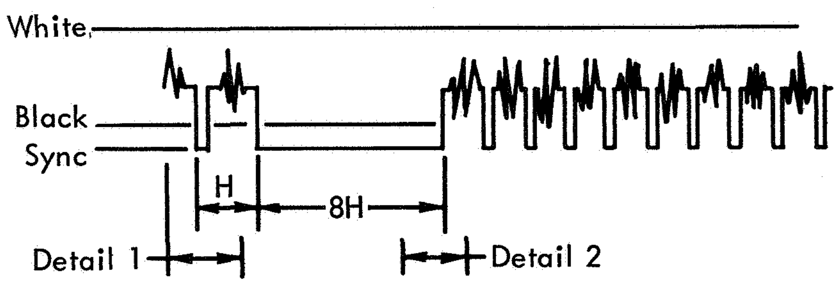
The equipment provides image tube surface scanning such that the aspect ratio of the finalized picture is 4:3. The scanned format yielding video is within the scene imaged on the vidicon.

The equipment is capable of developing a 2 plus 0.1, minus 0.6 volt peak-to-peak video signal across a 100 ohm resistive load. The video output line impedance of the television camera is 95 ohms plus or minus 10 percent.

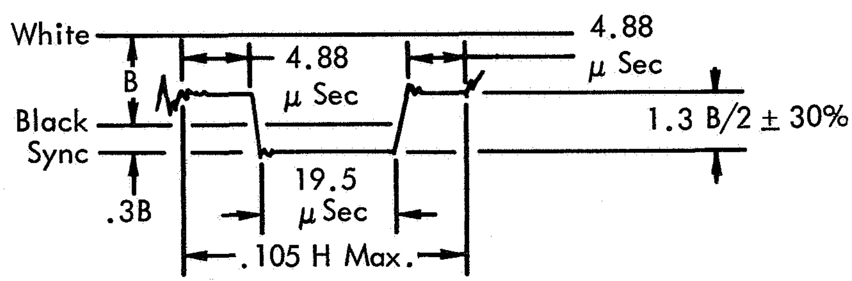
The equipment is capable of developing an output voltage as a function of specified input illumination of zero to 5 volts across a 1 megohm resistive load. The output impedance of this circuit is less than 5000 ohms.

The equipment develops white positive polarity across the resistive load defined.

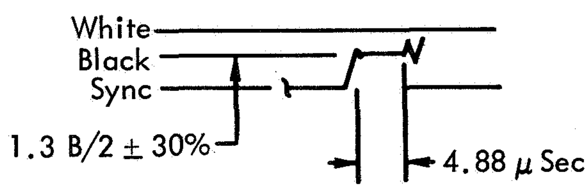
The equipment video output provides a peak-to-peak signal to root mean square noise ratio as listed below, in table 36 with the illumination levels specified herein.



Composite waveform



Horizontal blank and sync
Detail 1



Vertical sync and transition
Detail 2

Notes: Unless otherwise specified.

1. All dimensions nominal unless otherwise indicated
2. The rise time, being determined by a 500 KC low pass filter, shall be .8 μsec nominal
3. H = Line period = 312.5 μ sec
4. B = White to black amplitude
5. Porch duration (4.88 μ sec)

Figure 48. - Video wave form diagram

TABLE 36. - VIDEO OUTPUT SIGNAL REQUIREMENTS

Mission Phase	Environment	S/N
Bench checkout	Ambient	37
Boost	Specified operational design parameters	31
Spaceflight	Specified operational design temperature, pressure, pressure humidity and oxidation	34

Any periodic noise pulses such as power supply spikes that appear in the output video are limited in amplitude to twice the peak-to-peak value of the random noise level. Signal to noise shall be measured from the random noise level.

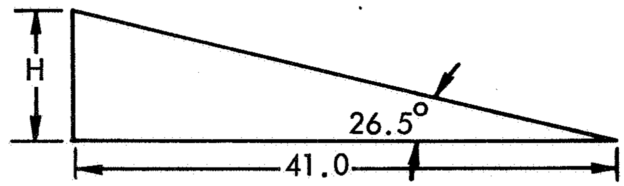
The optical system is a two lens turret composed of a standard Apollo 80° field-of-view (FOV), f2 lens and a qualifiable lens with 10° to 15° field-of-view, f2. The equipment is capable of performing with an incident highlight illumination on the vidicon photo conductor over a range of 0.1 to 30 foot candles. A 3 db reduction in the specified signal-to-noise ratio requirements may accompany a highlight illumination in the range of 0.1 to 0.5 foot-candle on the photo conductor. The video line spacing for both lens is shown in figure 49. To adapt the standard Apollo television camera for use in the Primate Spacecraft, modifications are required to bypass the power switch and probably to remove the pistol grip handle. Also, the lens mount will be removed and camera case modified to accept a power actuated, two lens turret. Some qualification testing will be required to prove the turret's integrity. The outline dimensions are shown in figure 50. The standard Apollo lens is shown, the narrow angle lens will be longer.

Within the Life Cell, the cage illumination source will be shaded from the camera optics such that no direct light is within the field of view. Also, the cage surfaces and equipment surfaces which are within the FOV will be prepared in a manner which will reduce highlights. A five-gray-scale test stripe will be included within the FOV for comparison with the primates' skin coloring. This will provide inorbit comparison which will eliminate any questions of camera drifts, equipment noise changes or down link noise.

Television integration within the Life Cell: The television subsystem is composed of two television cameras and one video recorder. The following modes of operation are planned:

(1) Transmit real time television from cell 1
(2) Record television from cell 1
(3) Dump television
(4) Record television from cell 2

1. With wide angle lens (80°)

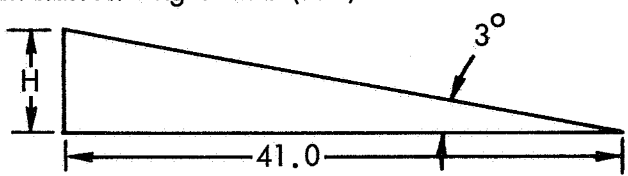


2H inches contains 250 lines (249 spaces)

$$\text{Line spacing} = \frac{2H}{249} = \frac{2(41.0)\tan 26.5^\circ}{249}$$

$$= 0.164 \text{ inches}$$

2. With narrow angle lens (10°)



$$\text{Line spacing} = \frac{2H}{249} = \frac{2(41)\tan 3^\circ}{249}$$

$$= 0.017 \text{ inches}$$

Figure 49. - Video line spacing

(5) Transmit real time television from cell 2

(6) For each of the above five functions there are two modes each, that is, lens turret position, wide angle lens or narrow angle lens.

The required system control will be provided by a television control module. This module will provide for turret drive control, selection of television camera, turn on the microphone for primate voice, and application of time annotation from the programmer sequencer. Television system activation and recorder turn on is per the time line figure 51. The Vidicon tube requires approximately 20 seconds warmup. This function will also be provided by the television control module by supplying camera power prior to recorder or transmitter power. Inputs to the television control module will be from the programmer/sequencer and the up-date link.

The turret will be mechanized such that the wide angle lens will be in place for the normal position, spring return to normal, and upon ground command/or feeder/waterer proximity switch signal. The power drive will rotate the turret to its narrow angle position as long as the command is retained. An override ground command will provide a disable of the proximity switch actuated command.

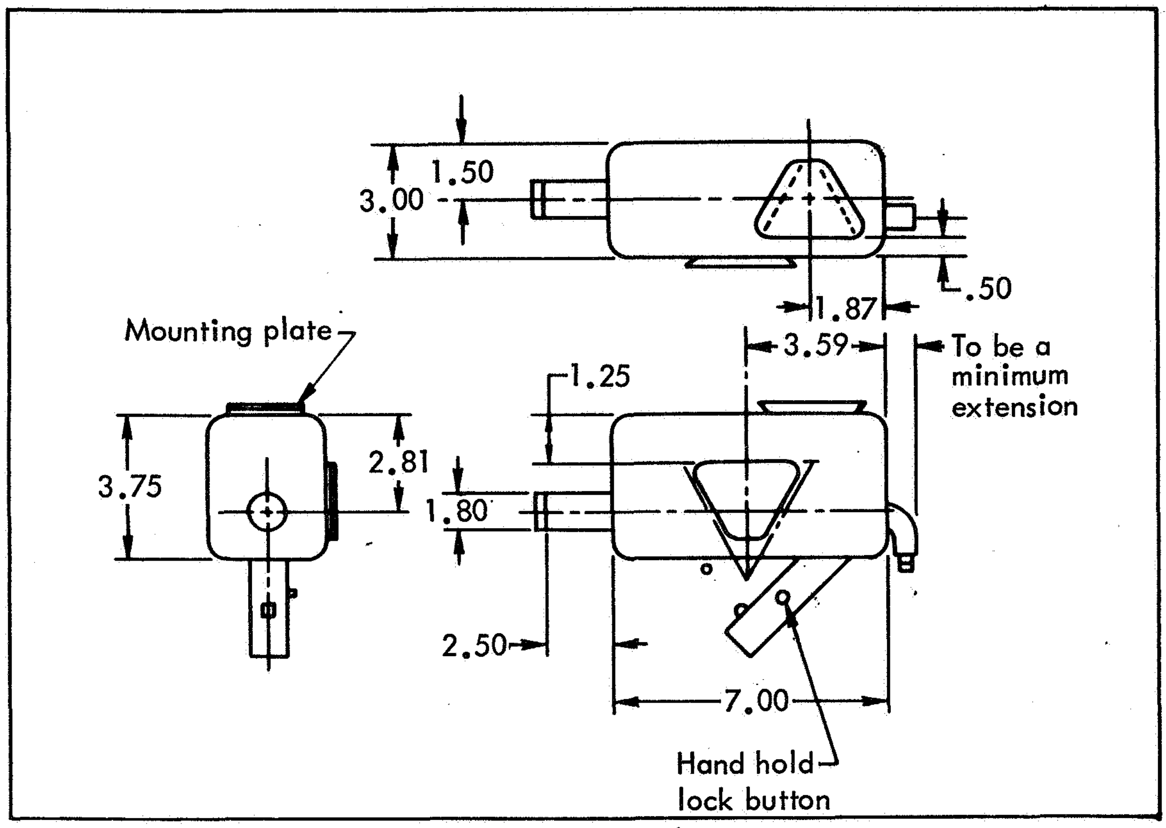


Figure 50. - Video camera outline dimensions

Provisions will be made for ground commands to select which cage is televised during real time and switching from one cage to the other will also be possible.

Upon passing out of ground control range, the turret will automatically return to normal position, wide angle, and a resumption of the time line functions will also automatically commence The Television System Block Diagram, as shown in figure 52.

During the fourteen hours of simulated daylight, the life cell illumination system is required to provide a minimum of 25 foot-candles of illumination in the life cell. A preliminary life cell configuration is shown in figure 53. In order to simplify the illumination design task, the life cell configuration of figure 54 was assumed.

The walls of the life cell were assumed to be reflective and have perfect diffusion, resulting in a reflection factor of 0.7. This is approximately the reflection factor for polished aluminum, porcelain enamel, or dull white paint. Reflection factors of 0.9 are obtainable using mirrored glass or bright white paint.

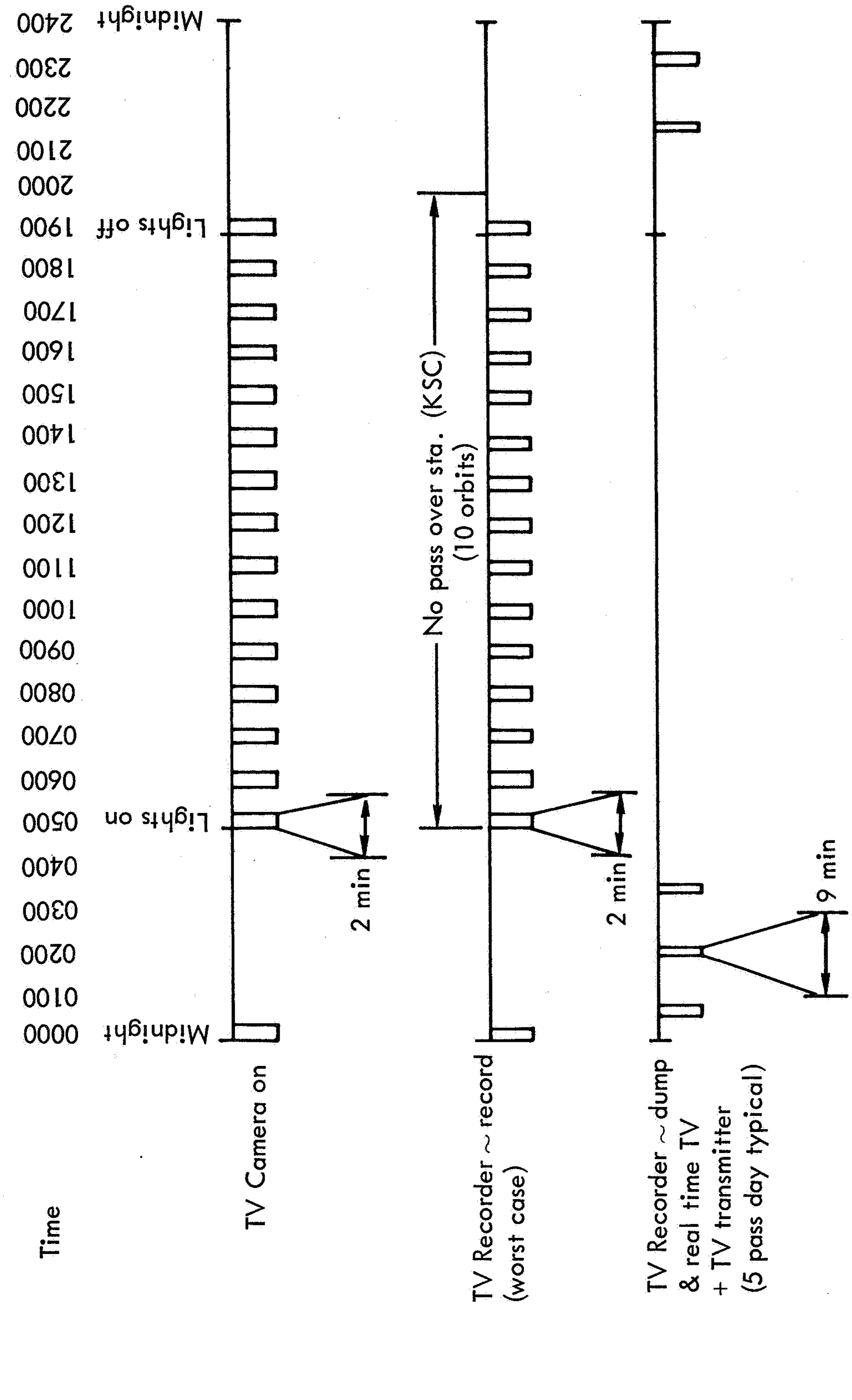


Figure 51. - TV time line

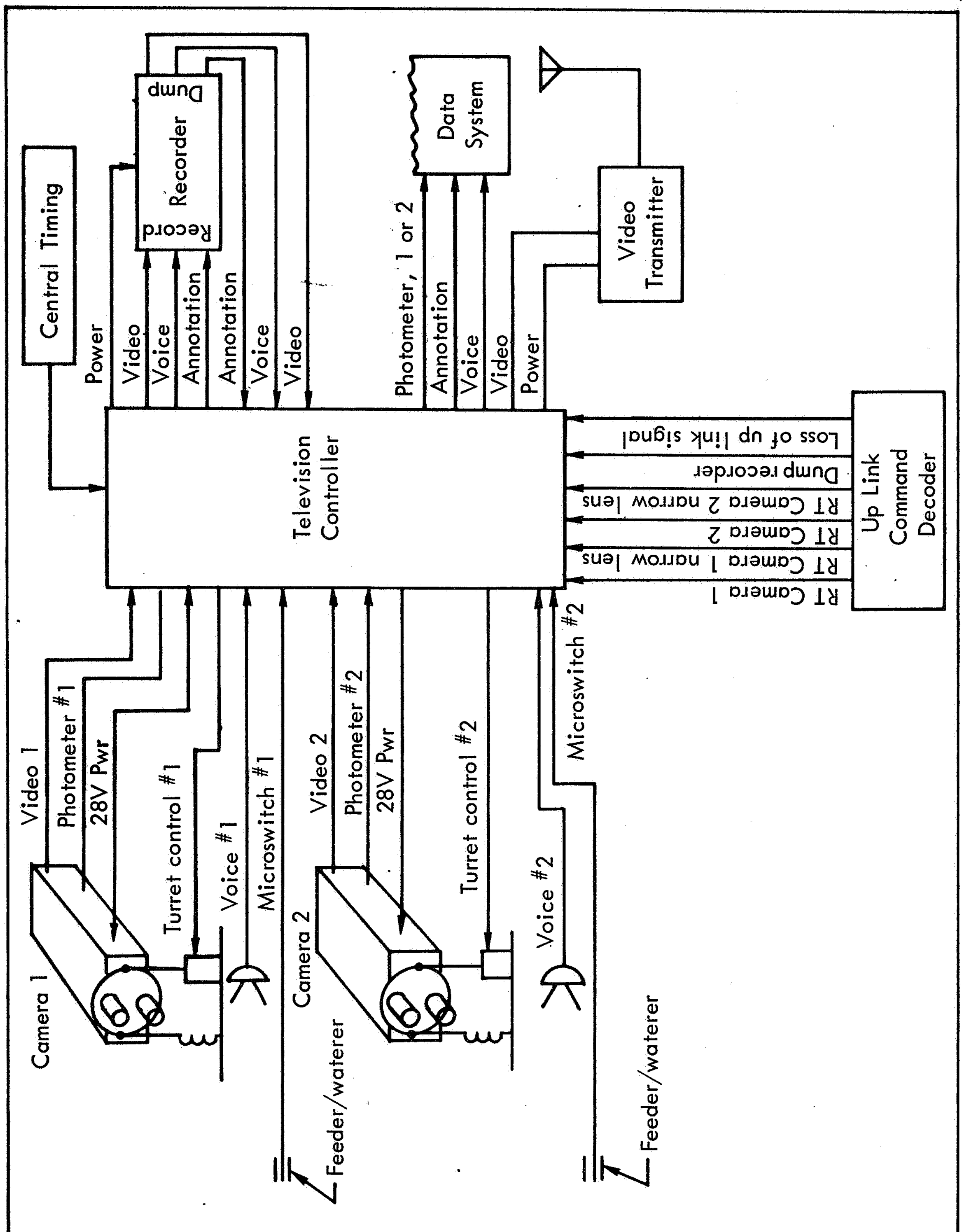


Figure 52. - TV mechanization block diagram

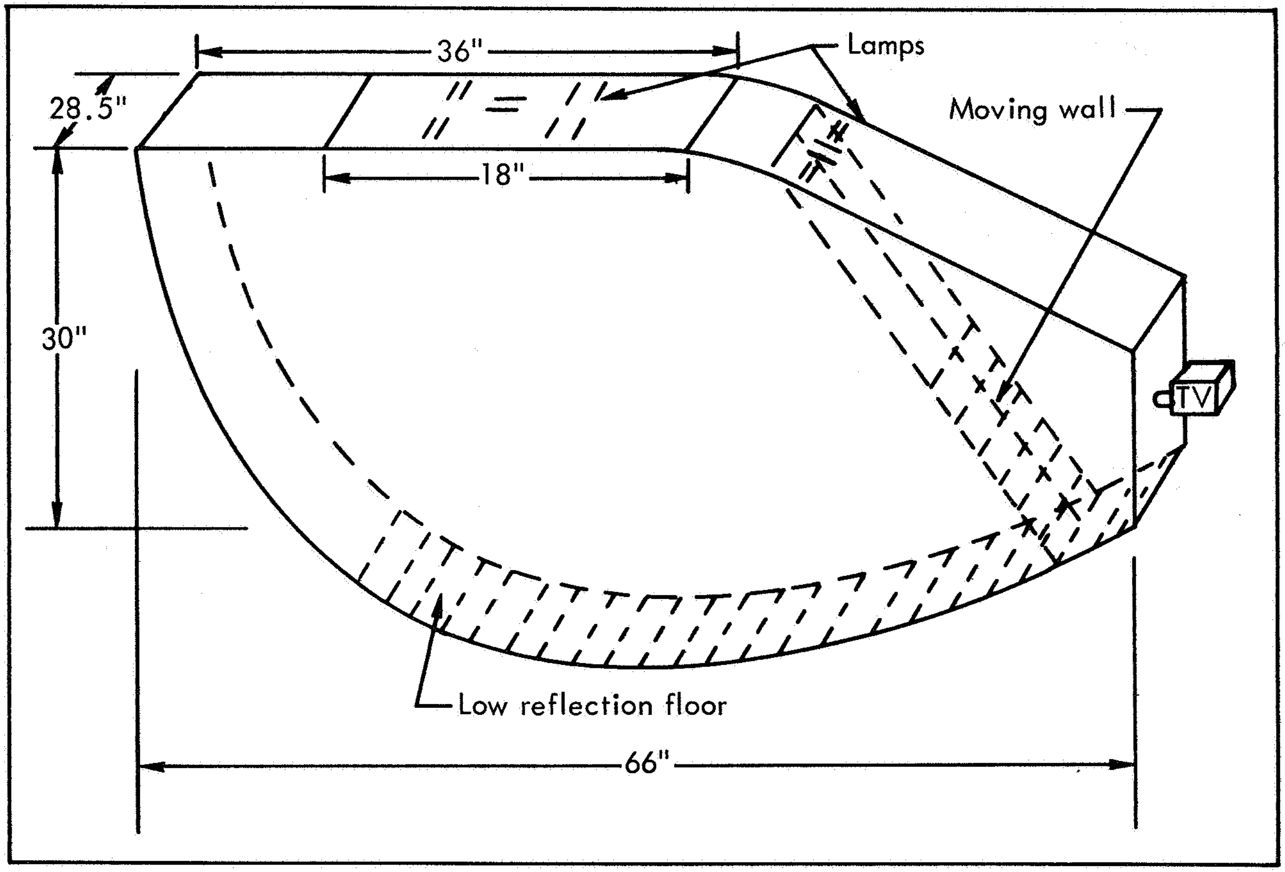


Figure 53. - Life cell configuration

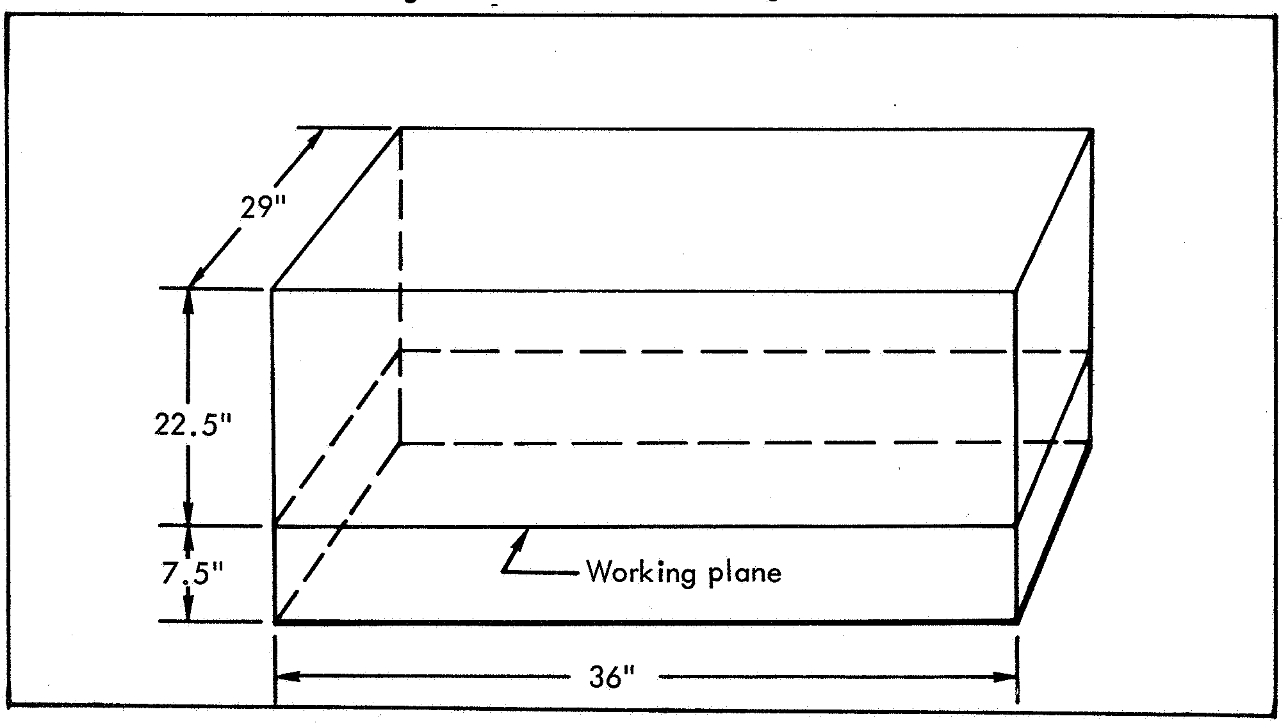


Figure 54. - Approximate life cell configuration

An average wall reflection factor ρ_w of 0.5 was assumed for the simplified configuration. Ceiling and floor reflection factors, ρ_c and ρ_f, were assumed as 0.7 and 0.1 respectively.

The room coefficient (k_r) is:

$$k_r = \frac{h\ (\ell + w)}{2\ \ell\ w}$$

$$= \frac{30\ (36 + 29)}{(2)\ (36)\ (29)}$$

$$= 0.933$$

A plot of E/B_{oc} for data available is shown in figure 55.

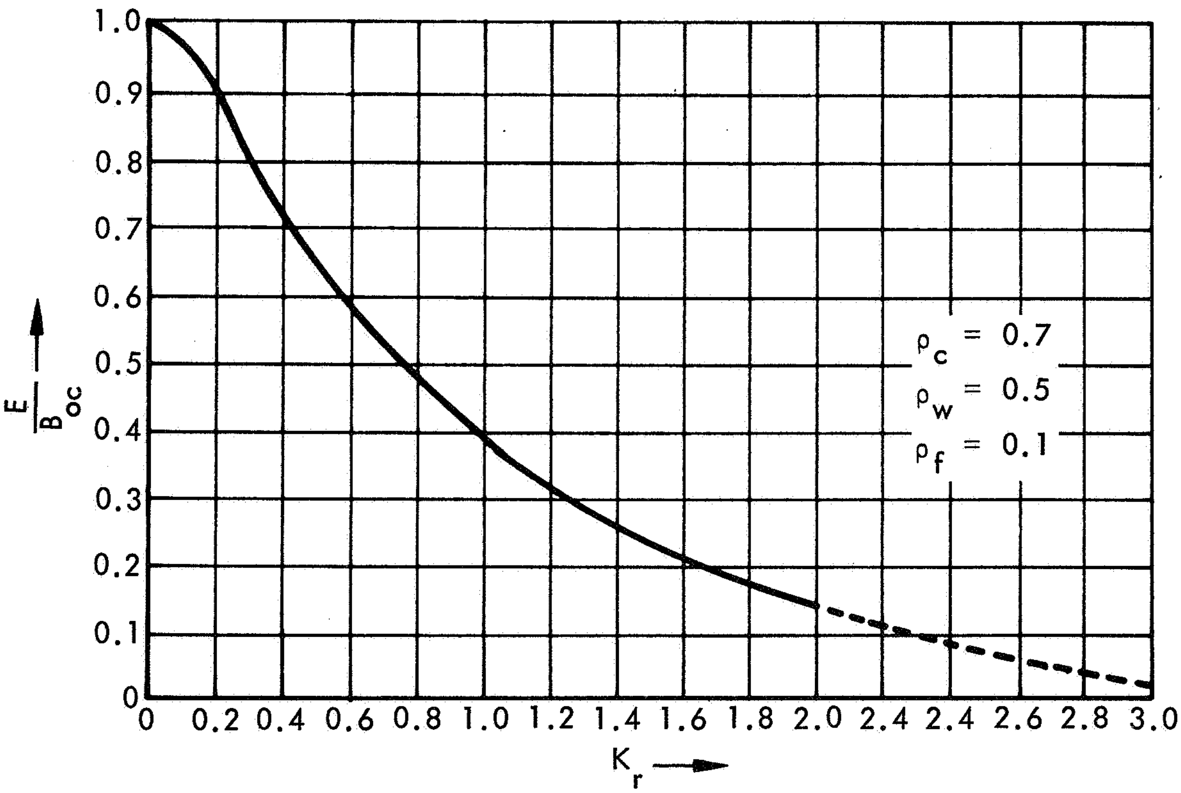


Figure 55. - Component ceiling lighting

For the working plane illumination, E, to be 25 foot-candles, the rationalized brightness, B_{oc}, of the ceiling must be:

$$\frac{E}{B_{oc}} = 0.5 \text{ (from figure 54)}$$

$$B_{oc} \cong \frac{25 \text{ ft-c}}{0.5} \cong 50 \text{ ft.-1}$$

The total ceiling flux, ϕ_c, for a 2.5 ft x 3.0 ft. ceiling is:

$$\phi_c = \frac{(50 \text{ ft-L})\ (2.5 \text{ ft})\ (3.0 \text{ ft})}{0.7} \cong 545 \text{ lumens}$$

Since the source plane is approximately 1.5 ft x 2.5 ft, the source plane brightness, B_p is:

$$B_p = \frac{545 \text{ Lumens}}{(1.5 \text{ ft})\ (2.5 \text{ ft})} \cong 145 \text{ ft-1.}$$

Assuming a luminaire panel transmission factor of 0.7, flux, ϕ_i, inside of the luminaire is:

$$\phi_i = \frac{(145 \text{ ft-L})\ (1.5 \text{ ft})\ (2.5 \text{ ft})}{0.7} = 780 \text{ lumens}$$

Assuming a luminaire efficiency of 80%, lamp lumens required, ϕ_e, are:

$$\phi_e = \frac{780 \text{ lumens}}{0.80} = 975 \text{ lumens}$$

To provide this quantity of lamp flux, eight six-inch T-5 florescent lamps and approximately 32 watts of power are required for each cell.

The simplified analysis of the illumination requirements and resulting power allotment was adequate for a preliminary design of the Primate cell. A more detailed analysis is required to determine the effect of configurational changes and the effect of different surface coatings as well as the effect of surface degradation.

An illumination network has been derived for the primate cell, the solution of which in an inverted matrix form permits a parametric study of illumination within the primate cell. Position and location of the primate may be varied, surface characteristics can be varied, and location of the illumination sources can be varied.

The shape factors from all surfaces within the primate cell were determined utilizing the CONFAC II computer program. The cell outline used was identical with the present cell design in dimensions and shape. The primate was modeled as two cylinders, 6 by 6 inches for the head and 9 by 12 inches for the trunk of the primate. Because of the transformation data within the CONFAC II program it is possible to locate the primate at any location within the primate cell and in any orientation.

The shape factors with the corresponding reflectivites were inserted into a radiation interchange matrix, and inverted on the 1620 IBM computer for two cases, the primate at a position just beyond the working panel and in the mass-measurement device. Two light sources were considered; one light source was behind the screen grid, the other light source was located on the ceiling. The results are tabulated below in Table 37.

TABLE 37. - IRRADIATION OF LIFE CELL SURFACES

CASE A ILLUMINATION BEHIND GRID
125 FOOT LAMBERTS AT GRID

Life cell interior surfaces	Primate in cell	Primate in M/M device
Side irradiation (P = .7)	30.4 foot lamberts	30.7 foot lamberts
Waste screen (P = .1)	34.1	37.0
Restraint grid (P = .5)	14.0	14.0
Top of cell (P = .7)	22.0	21.9
M/M device (P = .7)	14.7	14.4
Onto primate (P = .5)	17.8	14.4
CASE B ILLUMINATION OF TOP OF CELL 250 FOOT LAMBERTS AT TOP		
Side irradiation	32.4 foot lamberts	32.8 foot lamberts
Waste screen	30.4	34.6
Restraint grid	21.9	21.8
Light plane at top	22.0	21.0
M/M device	13.6	13.0
Onto primate	27.6	(13.0)

In order to determine the effect of variations in surface properties, the computer solution was extended by means of an equivalent electrical network of the radiation interchange matrix. By so doing, it is possible to separate out the effects of geometry from those of surface properties. The effect of wall reflecting is illustrated in figure 56.

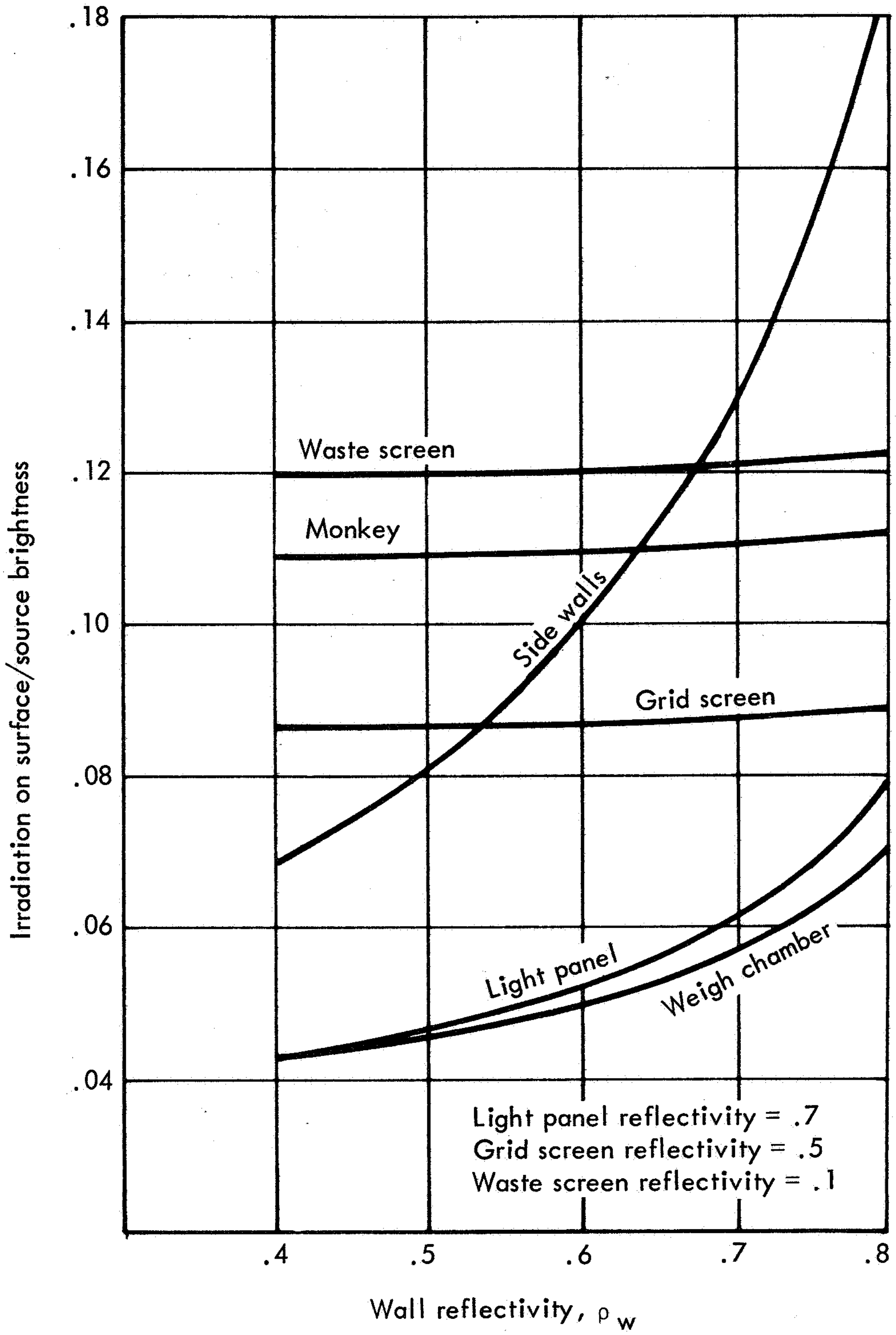


Figure 56. - Effect of wall reflectivity

The fouling of the waste screen may have the effect of increasing or decreasing the reflectivity of the waste screen surface. This effect is summarized in table 38, which tabulates the ratio of irradiation to ceiling source luminosity. It is evident that with extreme values of expected reflectivities between 0.1 and 0.25 the irradiation is only modified slightly. For this case, unit illumination is assumed at the ceiling and wall reflectivity is assumed as 0.7.

TABLE 38. - EFFECT OF WASTE SCREEN REFLECTION UPON IRRADIATION OF LIFE CELL

Life cell surface reflection	Reflection ratios			
Waste screen reflectivity	.1	.15	.20	.25
Irradiation of sides	.130	.131	.131	.131
Irradiation of waste screen	.122	.128	.136	.144
Irradiation of grid screen	.088	.088	.088	.089
Irradiation of top of cell (light)	.062	.065	.069	.074
Irradiation of M/M device	.057	.057	.057	.057
Irradiation of monkey	.111	.111	.111	.112

In a similar manner it was possible to determine the effect of ceiling louver characteristics. A transparent louver and a diffuse louver were assumed For the transparent case, a glass louver was assumed, having a transparency of 0.8 and reflectivity of 0.1. For the diffuse case, a reflectivity of 0.4 and transmissivity of 0.5 were assumed. In both cases the reflectivity of the fixture surface above the glass was assumed as 0.8. Wall reflectivity was assumed as 0.7, waste reflectivity 0.1 and grid reflectivity 0.5. The results are tabulated in table 39.

TABLE 39. - IRRADIATION OF PRIMATE CELL (Source 20 watt T12 lamp)

	Transparent louver ft. lamberts	Diffuse louver ft. lamberts
Wall irradiation	27.0	17.6
Waste screen irradiation	25.3	16.5
Grid screen irradiation	18.3	12.0
Weigh chamber irradiation	11.3	7.4
Primate irradiation	23.0	15.1

The penalty for using a non-glare type of fixture is a significant reduction in irradiation. It is not believed that the glare will significantly interfere with system performance.

From this study, it has been concluded that the results of the simplified analyses are substantially correct. The illumination will be adequate with the present design using 32 watt fluorescent lamps for each cell. These must, however, be located at the ceiling, and the lighting fixtures must be of a transparent rather than a diffuse type. Fouling of the waste screen will not substantially modify the luminous environment although fouling of the lighting fixtures will be a very significant factor. These factors, in addition to actual reflectivities and luminous outputs, must be confirmed by test.

Biotelemetry receiving: The biotelemetry receiving equipment will be composed of three items: receiving antennas, superheterodyne radio receiver, with a wideband output, and subcarrier filters and demodulator (discriminators).

The antenna system is composed of three, orthogonal antennas, tuned to the carrier frequency. The cage size will limit the antenna length and therefore the antenna may require lumped constant tuning to provide good impedance match to the coax transmission line and receiver, depending on the carrier frequency to be used. In Trade Study 3.2.2.3 (ref. 6) it is shown that the antenna can be considered in the far field for the short antenna that will fit in the cage (1/2 meter long).

The required receiver sensitivity is approximately -77.5 dbm for a 50 MHz carrier and -99.5 dbm for a 200 MHz carrier. This is based on the value of -50 dbm for the implanted transmitter. To obtain this required sensitivity a superheterodyne receiver will be required. The video, IF, output will provide the input to filters to separate the subcarriers for ECG, respiration and body temperature. These subcarriers will be demodulated in three discriminators and buffered to provide 0 to 5 volt signals to the telemetry system.

A diversity switching system will be provided to select the best signal to noise ratio receiver video output to the filter-discriminators, figure 57.

As stated above, the biotelemetry system uses three radio receivers and three associated antennas orthogonally placed within the cage to prevent cross polarization, thereby insuring a continuous signal. Any position change of the primates implanted transmitting antenna (probably consisting of only a tank coil) will affect the received signal strength. This change in signal strength may be very small or very large depending on the angular change with respect to each receiving antenna. By selecting a Schmitt trigger level, the required sensitivity may be obtained.

By ac coupling, only changes in the signal strength output are detected. A Schmitt trigger is employed to detect voltage level shifts in a consistent manner and generate pulses of constant amplitude. These pulses are counted in a digital counter, resulting in a count which, when read out periodically, indicates primate activity.

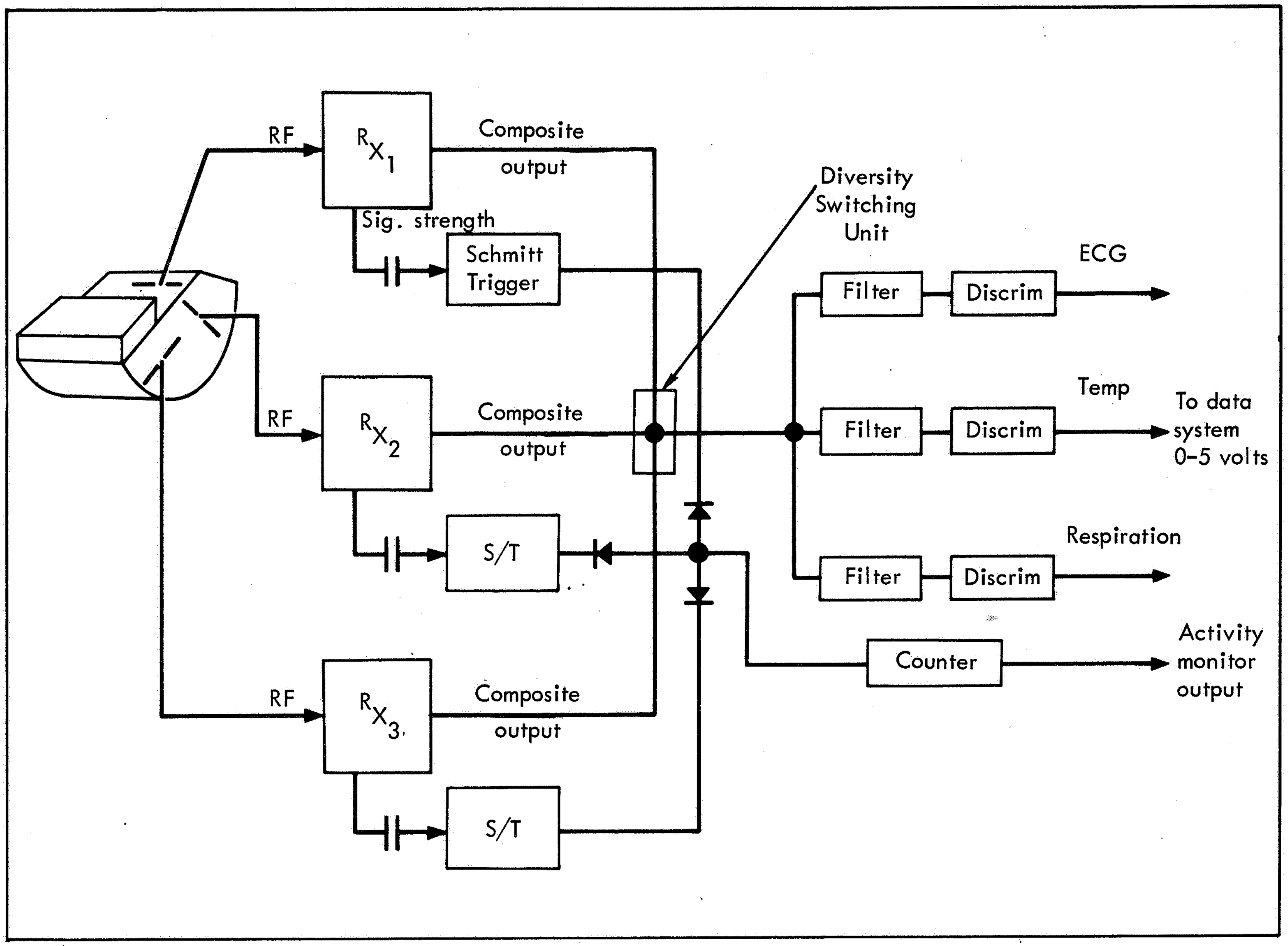


Figure 57. - Biotelemetry receivers block diagram

Since three antennas and receivers are used in each cage, the signal strength output from each receiver drives a separate Schmitt trigger. This allows for different antenna gains and receiver sensitivities to be adjusted out of the system. The three Schmitt triggers, with their ac coupled inputs, have their outputs diode summed. This provides a single input for the counter circuitry.

The counter and shift register will contain 7 flip flops for a maximum count of 128 motions in a one-minute sampling period.

Time annotation will be provided by the telemetry system in one-minute increments.

The Activity Monitor block diagram is shown in figure 58 and a typical signal strength diagram is shown in figure 59.

Dosimeter: This section describes the scintillation counter, or X-ray detector, developed and space flown on the OV1-2 Satellite by Northrop for the Air Force Weapons Laboratories. The research and development work was accomplished successfully by Northrop Systems Laboratories and a total of 18 months of orbital flight has been achieved on the FESS radiation measurement instrumentation. Only the X-ray detector will be utilized out of the FESS space proven radiation instrumentation to provide both proton and gamma dosimetry for the Orbiting Primate Spacecraft application.

The X-ray sensor element consists of a plastic scintillator (NE-103) mounted on a cesium iodide scintillator crystal, which is then cemented to a photomultiplier tube. The plastic is used to detect protons, giving rise to a fast pulse. The X-rays do not lose much energy in plastic, and pass unimpeded into the cesium iodide. The X-rays will give up a large amount of their energy to the cesium iodide, which has a high efficiency for stopping X-rays. This energy loss results in a pulse of light and hence a pulse from the photomultiplier tube. The current from the photomultiplier tube is then run through a wide-band dc amplifier to a digital converter. A block diagram is shown as figure 60.

As seen in the diagram, a logarithmic diode is used to obtain logarithmic expansion of four orders of magnitude for the photomultiplier tube and thus provide a range from one millirad/hour to ten rads/hour. The reference diode provides a stable base voltage for the differential amplifier against which any drift calibration corrections can be made.

The digital converter consists of an analog to digital converter which converts the 0 to 5 volt dc output of the dc amplifier into a six bit binary word. The digitized dose rate is then read out of the analog to digital converter into the dosimeter counter once per second by the telemetry master clock.

Thus, the dosimeter counter performs the integration function and accumulates the dose rate against time to give the total daily radiation dose in millirads/day. By varying the clock sampling interval, the dosage may be accumulated in either millirads or rads, as the experimenter desires.

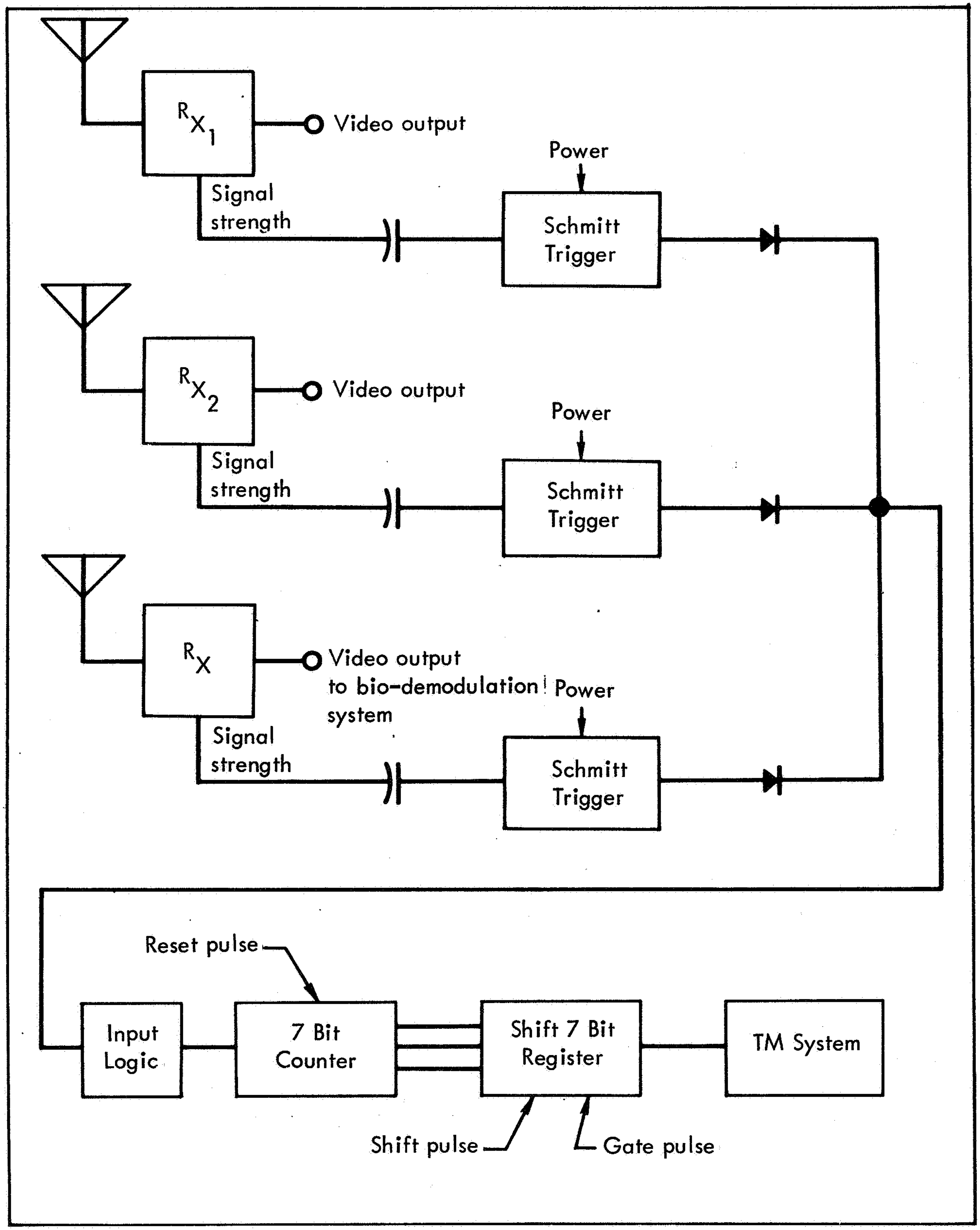


Figure 58. - Activity monitor block diagram

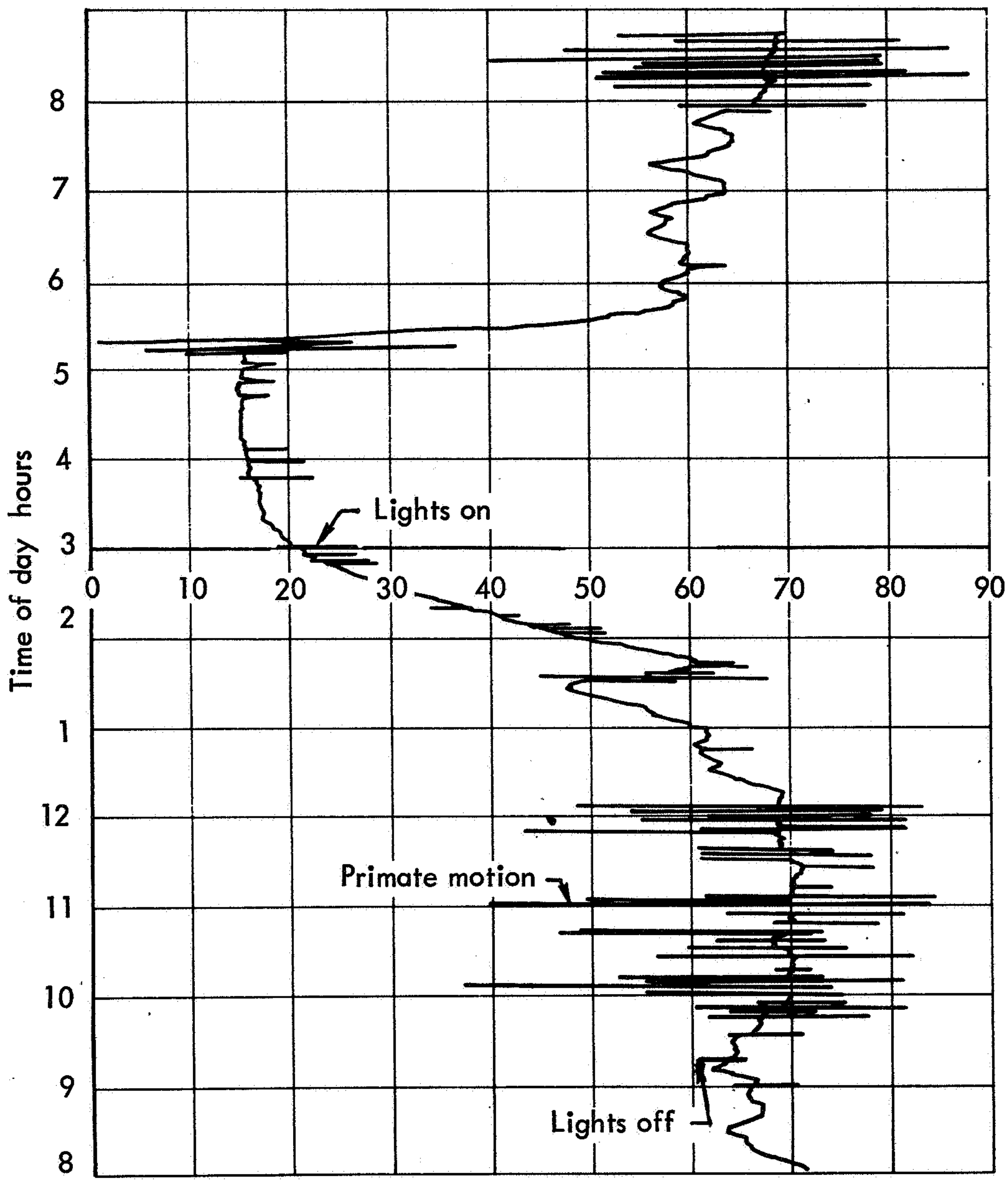


Figure 59. - Biotelemetry signal strength

The X-ray Detector is designed to measure the energy per unit time which bremsstrahlung from electrons striking the shielding of the instrument will produce in a thin cesium iodide crystal. Specifications are shown by table 40.

The scintillator assembly is cemented to an RCA 4439 photomultiplier tube with a solithane cement. Solithane was selected for use after an extensive program had demonstrated its usefulness under the environmental conditions. The photomultiplier tube is coated with a silastic rubber, RTV-11, and surrounded with a metal shield which is maintained at the photocathode potential. The tube is then wrapped with several layers of teflon tape, and inserted into a

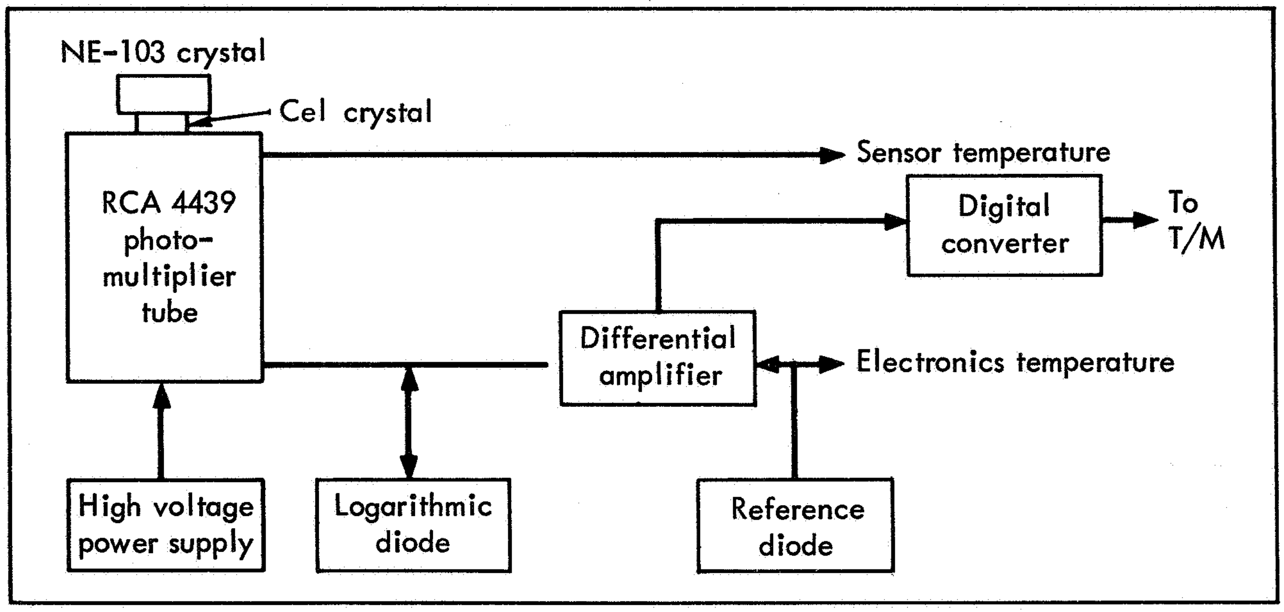


Figure 60. - Block diagram x-ray detector

TABLE 40. - X-RAY DETECTOR SPECIFICATIONS

Property	Value
Weight	2.20 pounds
Power	0.5 watts
Crystal	CsI (Tℓ) + NS-103
Sensitivity	1 millirad/hour to 10 rads
Telemetry	1 digital output, 16 bit word 2 analog temperature outputs

foam shell. The foam shell is then pressed into the 2-inch diameter aluminum sensor housing (shown in figure 61). The dynode components are mounted on a terminal board at the base of the tube, which is potted with silastic rubber, using the foam shell as a cup. The foam shell butts into a retainer ring which keeps it positioned in the sensor housing.

The high voltage supply for the photomultiplier tube is packaged in a self-contained can at the other end of the sensor housing. The supply consists of two printed circuit boards potted in the metal case. A lid is soldered on the end of the can with fingerstock insuring contact with the housing. The high voltage lead from the supply is connected to the photocathode lead through external microdot connectors.

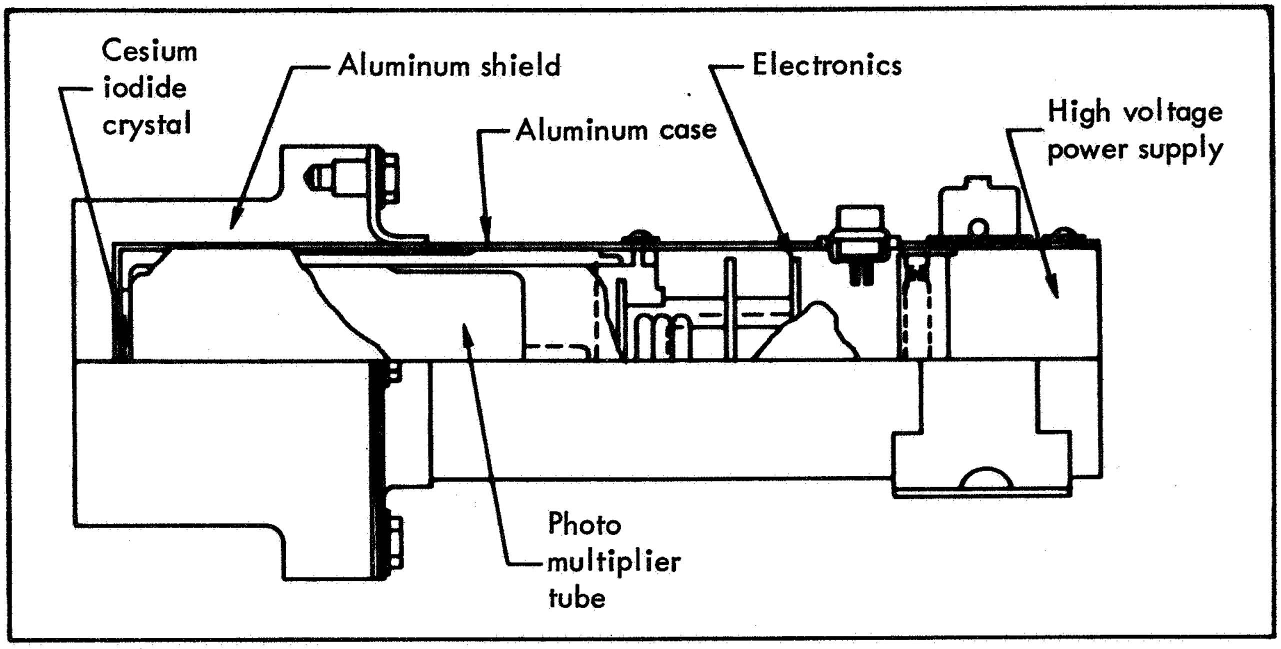


Figure 61. - X-ray detector sensor assembly

The signal leads in the X-ray detector and the circuit ground, are bought out of the sensor housing by microdot connectors and/or nine-pin connectors. The input power is supplied through the nine-pin connector, from the junction box.

The sensor shields are designed to minimize the radiation background and to define sensitive apertures for the instruments. The X-ray detector shield is entirely of aluminum.

In order to calibrate the X-ray detector, the energy flux striking the cesium iodide crystal was calculated for peak flux in the inner Van Allen belt. This was estimated as 1.08×10^5 Mev/sec. The photomultiplier tube saturates near 100 μa; thus the maximum current calculated for the peak flux was limited to 20 μa to provide a factor of 5 for safety. The calibration factor to be derived is then

$$F_{calc} = \frac{1.08 \times 10^5}{20} \frac{\text{Mev}}{\mu\text{a sec}} = 5400 \frac{\text{Mev}}{\mu\text{a sec}}$$

The calibration procedure is to irradiate the detector with a 1-mc cesium-137 source, the crystal is shielded by 5 gm/cm^2 aluminum. Only the cesium-137 gamma and bremsstrahlung will be present inside the shield. A multichannel analysis is made of the anode pulses from the photomultiplier tube, and the anode current is monitored. The energy calibration per channel was done by identifying the 662 Kev cesium-137 gamma peak, 3.28 Kev/channel, and calculating the area under the spectrum, 1.02×10^8 counts channels. Therefore the total energy flux seen by the crystal is, for a time of 600 seconds, $E_T = 557.5$ Mev/sec with the anode Ia, current measured, being 0.120 a.

$$F_{meas} = \frac{E_T}{I_a} = 5483 \text{ Mev}/\mu\text{a sec}$$

If the measured F does not equal the calculated F, the high voltage could be adjusted to yield the correct measured F. This is done by constructing a curve of anode current versus high voltage, as shown by figure 62. The ratio of F_{meas}/F_{calc} determines the current Ia required to yield the correct factor, and the high voltage is adjusted to yield this current, from figure 63.

The anode current is converted to a dc voltage output by the appropriate electronics, and a representative plot of this data at room temperature is shown in figure 63.

Data Processor: The Data Processor consists of logic circuitry coupled to binary registers and accumulative counters which convert event pulses from microswitches in the Behavioral Panel, Feeder, Waterer, and Exerciser into binary words representing start/stop times to total elapsed times. The driving clock pulses are obtained from the central timing unit of the Programmer and Sequencer.

The Behavioral Panel supplies the primate with a reward of food or water upon satisfactory completion of various tasks. The amount of work required to perform a given task is programmable and can be selected by ground command. Also, free food or water can be made available by an override feature initiated by up-link commands. To facilitate this description, reference is made to figures 64 and 65.

With the exception of timing and override features of the programmer sequencer, the Behavioral Panel electronics is self-motivated regarding stimulus and performance monitoring of the panel. Task selection circuitry selects the task to be performed which initiates the stimulus and task lamp. The Time and Responses circuitry monitors and prepares for storage the time from onset of stimulus to completion of the task. Provisions to compare the selected task with the task handle activations, and thereby determine if the task was performed properly or improperly, are illustrated. Data regarding the attempts at the task are time-annotated and prepared for storage. If the task requires a repeat performance due to errors, the repeat task circuitry informs the task selector, which repeats the onset of stimuli and task lamp. For a properly performed task, water or food reward is made available to the primate. Override by ground command is possible for repeated errors. Onboard programming provides a scramble sequence for presenting the task. Ground command can change the scramble sequence and also override task requirements for food or water.

The preceding discussion involved one primate and the Behavioral Panel. There are two independent panels, one for each primate. The task employing the

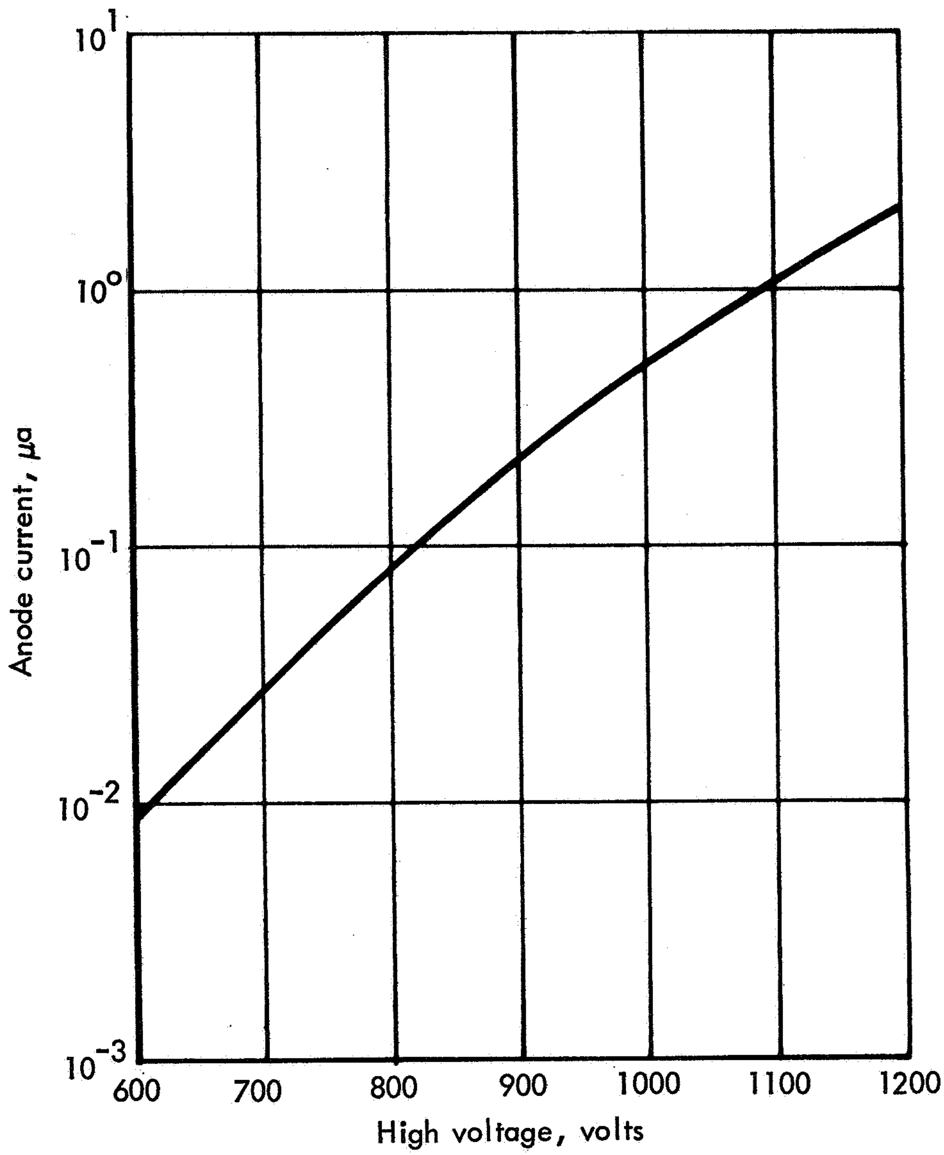


Figure 62. - Final calibration x-ray flight unit

audio click stimulus is also independent for each primate. The startle response is applied simultaneously to both primates due to their close proximity. More detail is presented in figure 64 regarding sequencing and collecting of data. The purpose of the circuitry is to time-tag, within 10 milliseconds, various tasks performed by the primates. Also, reaction time of the primate from onset of stimulus until completion of the task is time tagged. Provision to scramble task sequencing is provided and monitoring to insure correct responses have been performed. Should a primate refuse to work for reward, override by ground command will supply free food or water. Data is collected on as occurs basis, stored and dumped to the ground site with other data.

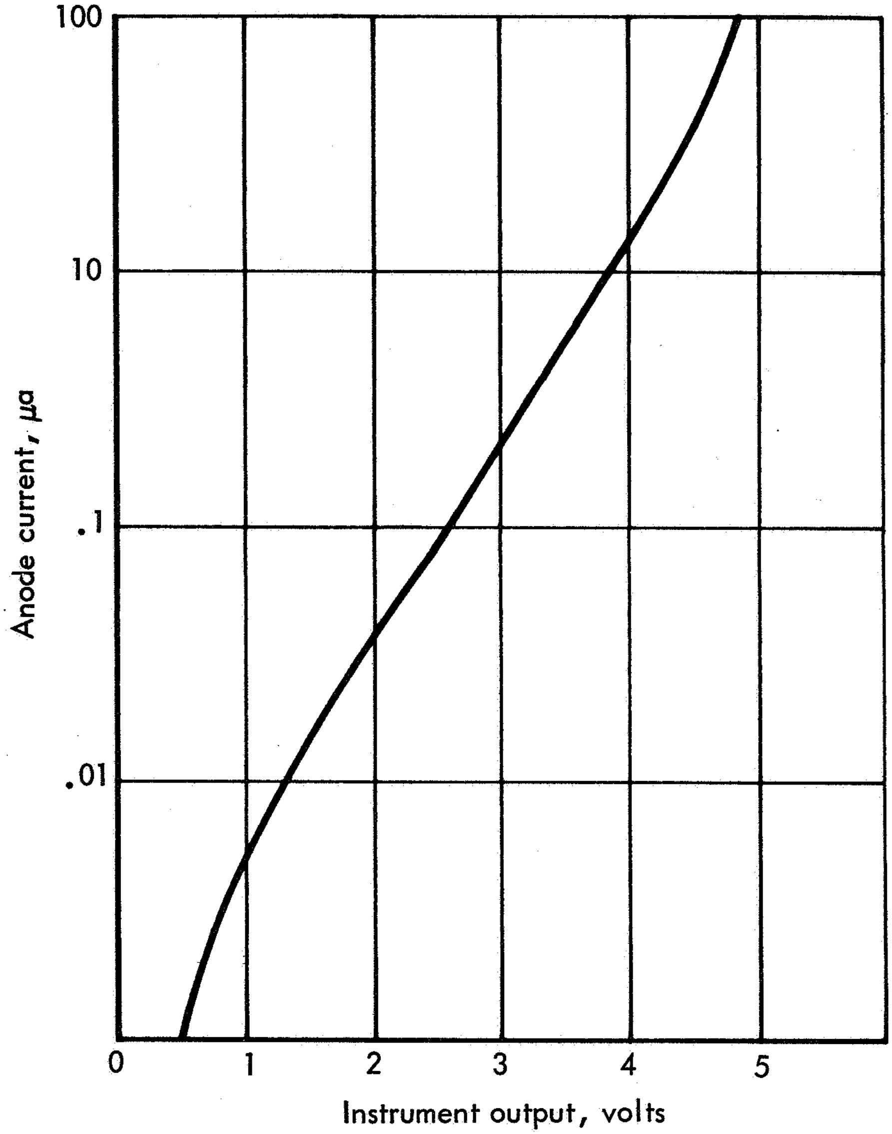


Figure 63. - Calibration flight of x-ray unit at +80° F

Various task data are sequenced, time annotated and stored in the temporary storage. When a sufficient amount of data is accumulated in the buffer storage, it is transferred to the magnetic tape recorder. Selection of a buffer storage requires that several assumptions be made regarding the data produced by each primate.

(1) No more than one task can be performed within one second.

(2) Maximum responses of 10 per second.

(3) Maximum bits per response is 12 (4 identification and 8 time bits)

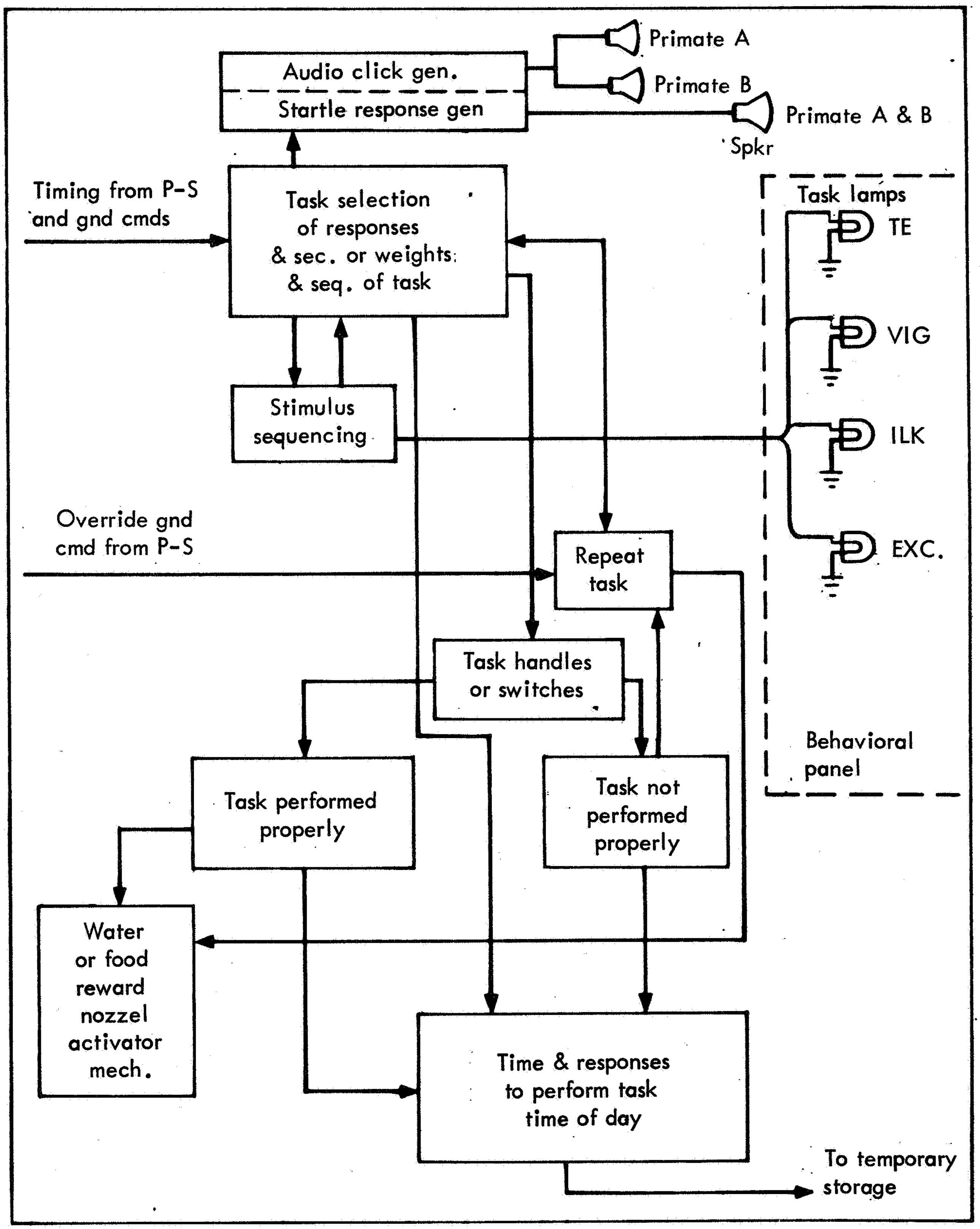


Figure 64. - Behavioral panel and data processor block diagram

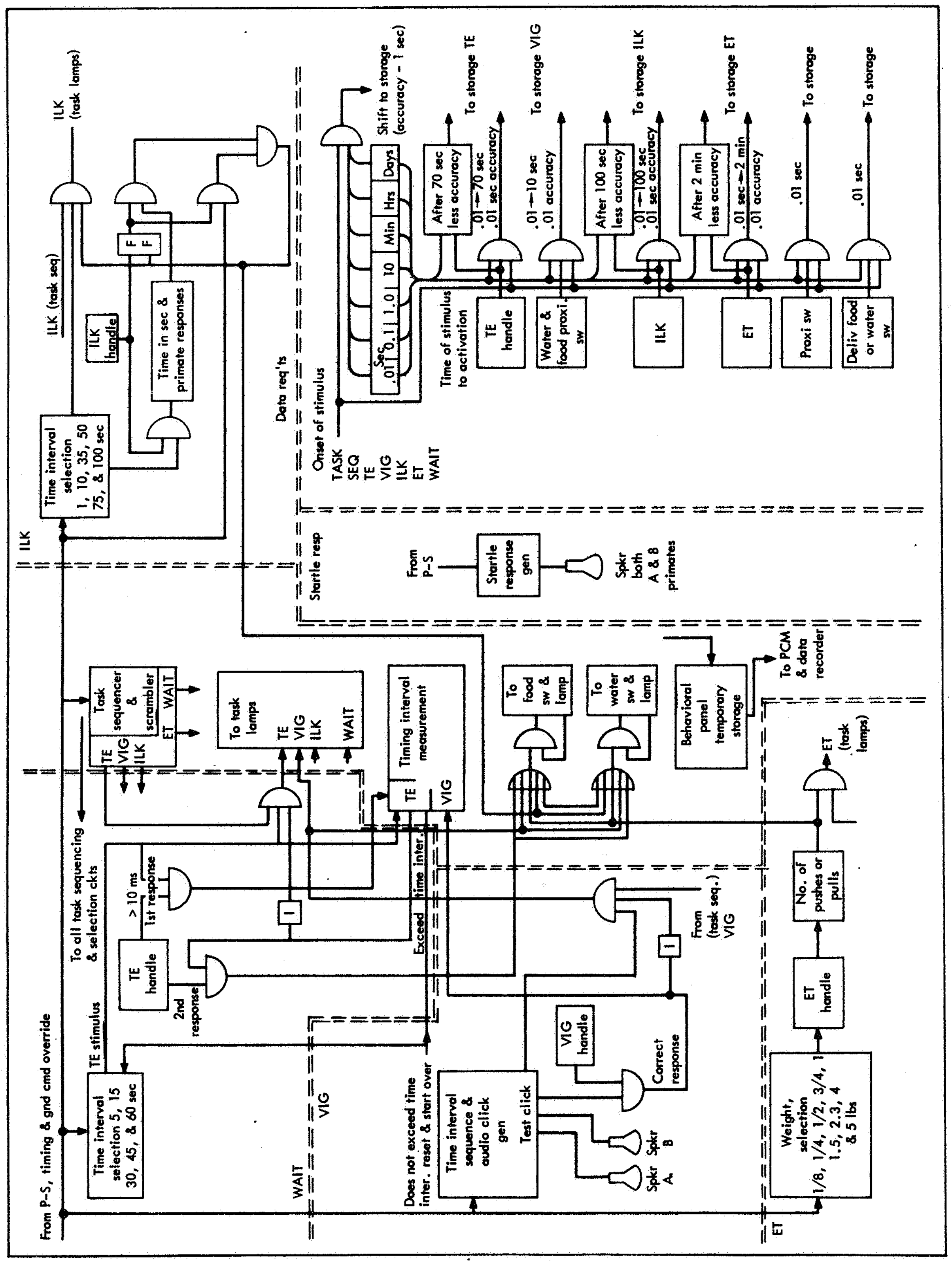


Figure 65. - Behavioral regime electronic block diagram

(4) Maximum bits that can occur during one second is 120 bits or 240 bits for both primates.

(5) Sufficient storage should allow a 5-minute recording of an EKG without interruption. Therefore, the storage should handle 5 minutes of behavioral panel activity at the highest rate of 120 bits per second. This requires a storage of 36×10^3 bits for each primate.

(6) This large quantity of bits is worst case and could be decreased through design techniques. A reduction in bit requirement should be realized in the method of time annotation.

The output of the waterer is a switch action energized by the primate's mouth which activates logic circuitry in the data processor to meter water flow. Input to activate the water nozzle is a successful completion of required tasks, or a ground command to override task performance which enables logic circuitry in the data processor to permit water flow. The feeder requires the same inputs for activation as the waterer. A food pellet is delivered through the mouthpiece upon proper completion of a task and placement of mouth around the nozzle, enabling logic circuitry in the data processor. Performance of a task will provide either food or water; the primate selects the desired one.

The data processor contains the analog-to-digital converter circuitry for the dosimeter, as well as the accumulative dosimeter 16-bit counter. Samples of the analog 0 to 5 volt dc dosimeter dose rate are obtained each second, digital-converted, and added to the total on the dosimeter counter. On readout command of the programmer, the total integrated dosage is read out into the digital multiplexer. The data processor acquires status events signals from switch closures or voltage impulses from the life cells and the various supporting spacecraft subsystems. These event signals are encoded by 8-bit binary registers into psuedo binary words wherein the bit location identifies the event and the one state signifies the event occurrence. The events binary words are read out of the data processor on the programmer command into the digital multiplexer and are interlaced into the digital data stream.

Spacecraft status panel: External on the spacecraft's outer shell and a adjacent to the recovery capsules, a Spacecraft Status Panel will be located to be clearly visible from the windows of the Apollo command module. The purpose of the status panel is to warn the astronauts of any major unsafe conditions existing in the Orbiting Primate Spacecraft prior to docking or prior to removal of the recovery capsule.

The Status Panel will consist of three high intensity incandescent lights of 90° beamwidth arranged in a triangle. The top or apex light will signify spacecraft unsafe condition by one-half second flashes. The right hand and left hand lights of the triangle, numbers 1 and 2 recovery capsules respectively, will signify capsule unsafe conditions by 2 second flashes. The light flashing system of the panel is relay driven by separate solid state logic circuits which are activited by the status signal outputs from the signal conditioner. The signal conditioner, in turn, derives its inputs from the spacecraft subsystems' transducers and sensors which measure the operational status on performance of each subsystem. By selecting the proper combination of sensor and transducer outputs to key the logic circuitry through voltage-level triggers, the status panel light can be keyed on in the flashing mode to indicate the unsafe condition desired.

Status Multiplexer: The status multiplexer monitors and sends to the ground station, signals of sufficient information to determine the operational status of the spacecraft. A simplified block diagram of the status multiplexer is shown in figure 66. A partial list of the data points are clock frequency, power supplies, and timing signals.

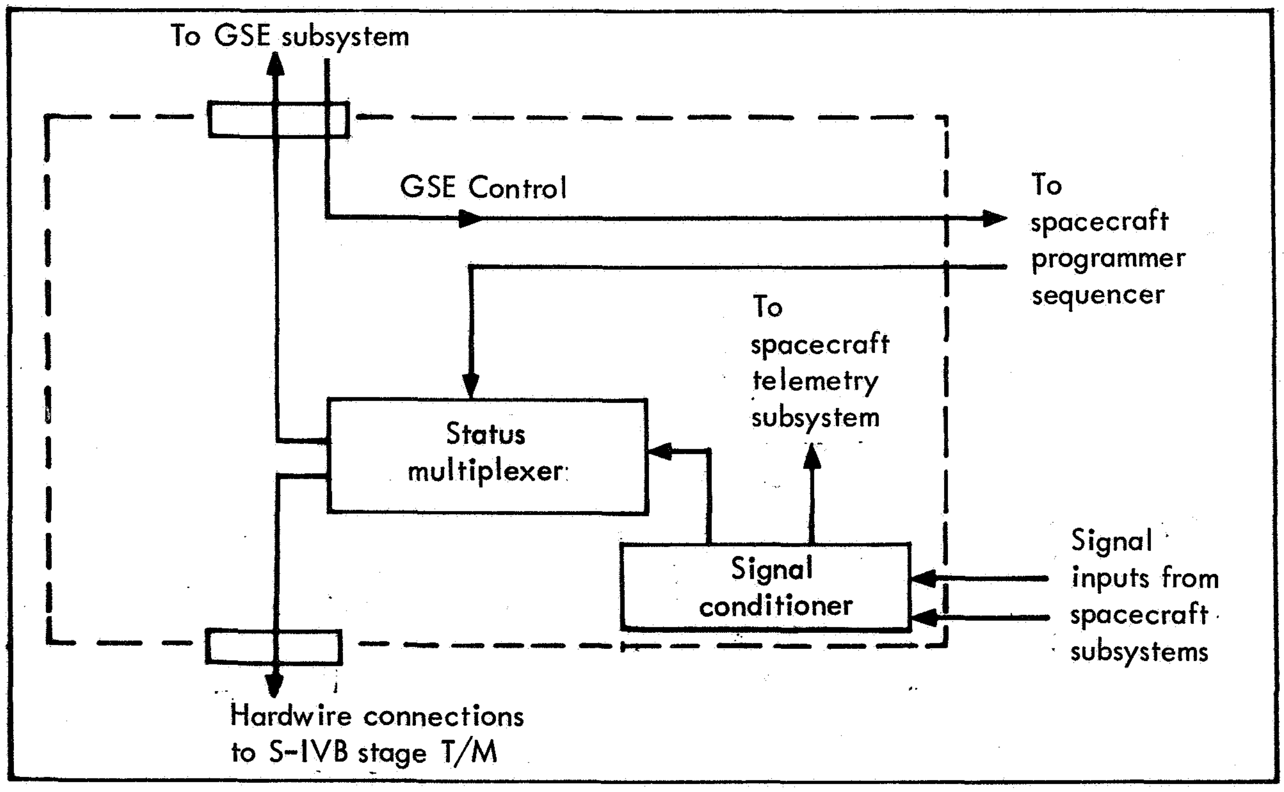


Figure 66. - Status multiplexer block diagram

Prior to lift-off, the status signals interface with the ground station through GSE equipment and connections. Prior to separation the status multiplexer commutates selected spacecraft engineering signals and provides S-IV-B telemetry with the resulting spacecraft status signals.

Advance development areas. - The following section discusses the instrumentation sub-area component possible development areas and suggests possible approaches. Upon conclusion of the laboratory test model testing, these areas will be redefined based on actual experimental results.

The development of the biotelemetry receiving antenna system will require simulation of the cage and the plotting of antenna patterns to establish adequate cage coverage. The impedance matching will be a method of increasing the electrical length of the short antenna. Depending upon the selected frequency range, the antenna type and loading technique will be determined.

For the activity monitor experiments will be required to determine the signal strength range and sensitivity required to provide an output pulse from the Schmitt trigger for a given amount of torso motion, and to obtain correlation between primate activity patterns and the activity count.

For the mass measurement device, due to the problems noted in this area it is recommended that a test program be instigated to develop a suitable weighing technique. The adiabatic volume measurement system is suggested as the most promising approach for preliminary testing. It does have some serious drawbacks however, which must be investigated. These are:

(1) Cooperation of primate in entering mass measurement device.
(2) Influence of primate's breathing upon accuracy.
(3) Confinement of primate in a small closed chamber.
(4) Contamination of the measurement device with debris.

Preliminary equipment list. - A preliminary list of equipment required in the Instrumentaties Subsystem is itemized in table 41.

TABLE 41. - INSTRUMENTATION SUBSYSTEM
PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	TV camera	RCA	Apollo camera	2
2	Biotel antenna	NSL	Special	6
3	Biotel receiver			6
4	Biotel filter			6
5	Biotel demodulator			6
6	Biotel diversity unit	NSL	Special	2
7	M^3 logic modules	NSL	Special	2
8	M^3 schmitt trigger	NSL	Special	6
9	Primate voice microphone			2
10	Primate voice compressor and buffer	NSL	Special	2

TABLE 41. - (concluded)

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
11	Experiment sensors table 33	Various		64
12	Engineering sensors table 34	Various		125
13	Signal conditioner assy.	NSL	Special	2
14	Dosimeter	NSL	Special	1
15	Data processor unit	NSL	Special	1
16	Astronaut safety status panel and electronics	NSL	Special	1
17	Status multiplexer	NSL	Special	1
18	Mass-measurement unit	NSL	Special	2

Mass measurement device: Determination of a body's mass in a zero gravity field is a difficult problem since weight forces by which masses are normally measured are totally absent. From basic physics we know that mass exhibits other properties which are amenable to measurement, independent of a gravitational field. Mass possesses the property of inertia, that is that a force is required to change velocity. Mass also occupies a finite volume, and may be measured volumetrically, if the density of the mass is known. Thus, it is possible to make mass determinations under zero gravity conditions, at least in the case of an ideal rigid-body homogenous mass.

Mass measurement of animate objects, such as the primates, is a problem, since these objects are far from the ideal case. Rigid-body assumptions are not valid and body volume is not homogeneous and is influenced by breathing. With man, these problems may be reduced with his cooperation by holding himself as rigid as possible for inertial measurement, or by holding his breath long enough for a volume measurement. In the case of monkeys, however, no such cooperation exists. He could move around, resist any attempt to restrain him, and even breathe harder due to fear or excitement.

Body weight changes are one of the more important indicators of the animal's physical condition. Excessive weight gain or loss would indicate a problem. Some control is possible by changing the amount of food available if the subject is taking food. If he is refusing food, then little can be done, but the weight data is still important to the physiologist.

Small variations in weight on an hourly or daily basis are normal. Definite weight trends can be ascertained from daily measurements plotted over a period of a week or so. Determination of early trends implies accurate measurement ±1% of body weight or better. If accuracies or sensitivity thresholds result in data which has more than ±2% error, then the subject must experience a large change before it can be detected. Errors as large as ±10% would be of little help to the physiologist, since the animal would already be in trouble before the problem is detected.

Thus, the weighing requirement may be summarized as follows: Measurement of weight, mass or volume, must be determined to 1% accuracy. However, absolute accuracy, including calibration traceable to NBS primary standards, is not a stringent requirement; ±10% is acceptable and the ±1% accuracy refers to the precision, threshold, and repeatability of the data on an end-to-end basis, including telemetry and ground detection and display equipment.

(1) Sample rate weighing is desired at least once per day. The weighing should not impose on acceleration force greater than 0.01 g.

(2) The amimal may be trained to enter a weighing device, if necessary, as a task function with stimulus cues and rewards to reinforce the task. The more simple and pleasnat the task, the greater the probability of gaining his cooperation.

(3) The animal cannot be required to constrain himself, stiffen his body, hold his breath, or otherwise behave in an unnatural manner.

The most promising technique to determin the weight or mass of the primate appears to be the adiabatic compression method of determining body volume. Body volume is an indirect measure of the animal's mass and may in some cases be a more meaningful indicator of gain or loss than weight itself.

The method consists of placing the animal in a closed, known volume chamber and measuring the compressible volume of the atmosphere surrounding the incompressible monkey. This may be accomplished by injecting a small known volume of gas into the container and measuring the small change in pressure produced both with the without the monkey inside the chamber. The magnitude of the change in pressure is directly proportional to the unknown gas volume, with the monkey's volume simply the difference between that and the volume of the container.

Of concern to the design of a practical adiabatic compression measurement system is:

(1) The pressure change must be very small about 0.1 psi to avoid physiological problems.

(2) The pressure change should be large .01 psi with respect to short-term pressure variations associated with animal respiration.

(3) The measurement must be made quickly to avoid thermal effects.

(4) The measurement should be sufficiently static to allow full compression of the subject's exhaled gases and thus, reduce respiration effects.

A device has been developed by Acoustica Associates, Inc., of Los Angeles, California, which appears to be applicable to the Orbiting Primate Spacecraft Program. This device employs an electromagnetically driven bellows which produces a cyclic displacement at 5 Hz. The use of a reference volume, with a capillary bleed port, assures static equilibrium and common gas properties of C_p C_v, P and T. Thus the unknown volume is simply a function of pressure change measured in the tank and in the reference volume.

Northrop performed tests using this device in an attempt to measure the volume of two macaca speciosa monkeys. Details of the test set-up and results are given in Trade Study 3.2.9.7.(ref. 6) A difficulty was encountered in that the volume reading oscillated and drifted over a $\pm$ 10% range, even though the device has a reported accuracy for liquids of $\pm$ 0.5%. This difficulty is attributed to animal respiration, since the cyclic rate is too fast to eliminate body dynamics of the chest cavity, and the pressure change was too low to eliminate respiration thermal effects. Northrop believes that a successful development could be carried out to optimize this system for the primate program with an accuracy appraching $\pm$ 2%.

Telemetry

The purpose of the Telemetry Subsystem is to receive command data from the ground, and transmit to the ground primate environmental data, primate physiological data, engineering data, and primate voice and television. This section provides a functional description of the Telemetry Subsystem design including the receiver, digital data encoding and storage, primate voice and video encoding data and television transmitters, and data and television antennas. Applicable performance data will also be described. Additional development effort has also been identified to insure availability of equipment within schedule constraints.

The Telemetry Subsystem coherently detects the up-link signal, tracks the up-link carrier and transmits down-link telemetry and voice as two subcarriers on a carrier coherently related to the up-link carrier. The down-link frequency is related to the up-link frequency by a fixed ratio of 240/221. Television data is transmitted by a separate transmitter on a carrier frequency which is different from the data/voice carrier frequency.

Voice and telemetry data are transmitted as subcarriers of 1.25 MHz and 1.024 MHz respectively. The carrier frequency is 2287.5 MHz. The television carrier frequency is 2272.5 MHz.

The approach presented herein evolved from the Tracking Network Optimization Launch Phase Data Transmission and Data Handling Trade Studies (ref 6).

Telemetry requirements and constraints. - The basic requirements of the Telemetry Subsystem are to receive command data from the ground and to transmit telemetry, primate voice and television to the ground. The Telemetry Subsystem is required to receive the up-link signal and detect and demodulate the 70-KHz up-link command subcarrier. The demodulated binary signal is sent to the command decoder for further processing. In addition, the telemetry system is required to track the up-link carrier in order to supply a reference for the down-link carrier and ranging code. The ranging code is used by the Mission Control Center, Houston, in trajectory computations.

It is required to transmit voice and telemetry on one transmitter and television on another transmitter due to the insufficient signal which would be received if only one transmitter were used for voice, television, and data. The Telemetry Subsystem is required to transmit approximately 200 channels of information. Due to the accuracy requirements of the data being transmitted, a PCM system rather than a PAM system is required. An eight bit word is used which gives a resolution of one part in 255.

The telemetry system is required to be compatible with the MSFN.

Telemetry subsystem functional description and performance. - Figure 67 is a simplified block diagram of the Telemetry Subsystem. There are two complete transponders, two television transmitters, and two voice/data transmitters including the necessary switching circuitry. For simplicity of presentation, redundant transmitters are not shown in the block diagram. To minimize new development and to assure compatibility with the MSFN, Apollo type equipment is selected wherever its usage is compatible with the primate spacecraft and mission. This equipment is listed in the Preliminary Equipment List and represents Apollo design and mechanization.

The receiver receives the up-link signal from the ground station. The up-link carrier frequency is tracked by a phase-locked loop to supply a reference for the down-link transmitted signal. In addition, the up data (command) subcarrier is demodulated by a tracking loop. The ranging code is derived by multiplexing, in the wide band detector, the rf input with the output of the VCO in the carrier tracking loop. However, the output of the VCO in the carrier tracking loop is an estimate of the phase of the input signal; therefore the wideband signal contains the ranging code.

The transmitted signal consists of a carrier which has telemetry data and voice subcarriers. The primate voice frequency modulates a 1.25 MHz subcarrier. The telemetry data, which is in digital form, bi-phase modulates a 1.024 MHz subcarrier. These two subcarriers phase modulate the 2287.5 MHz carrier.

The television signal frequency modulates a 2272.5 MHz carrier, which is transmitted independently of the voice data transmitter and antennas.

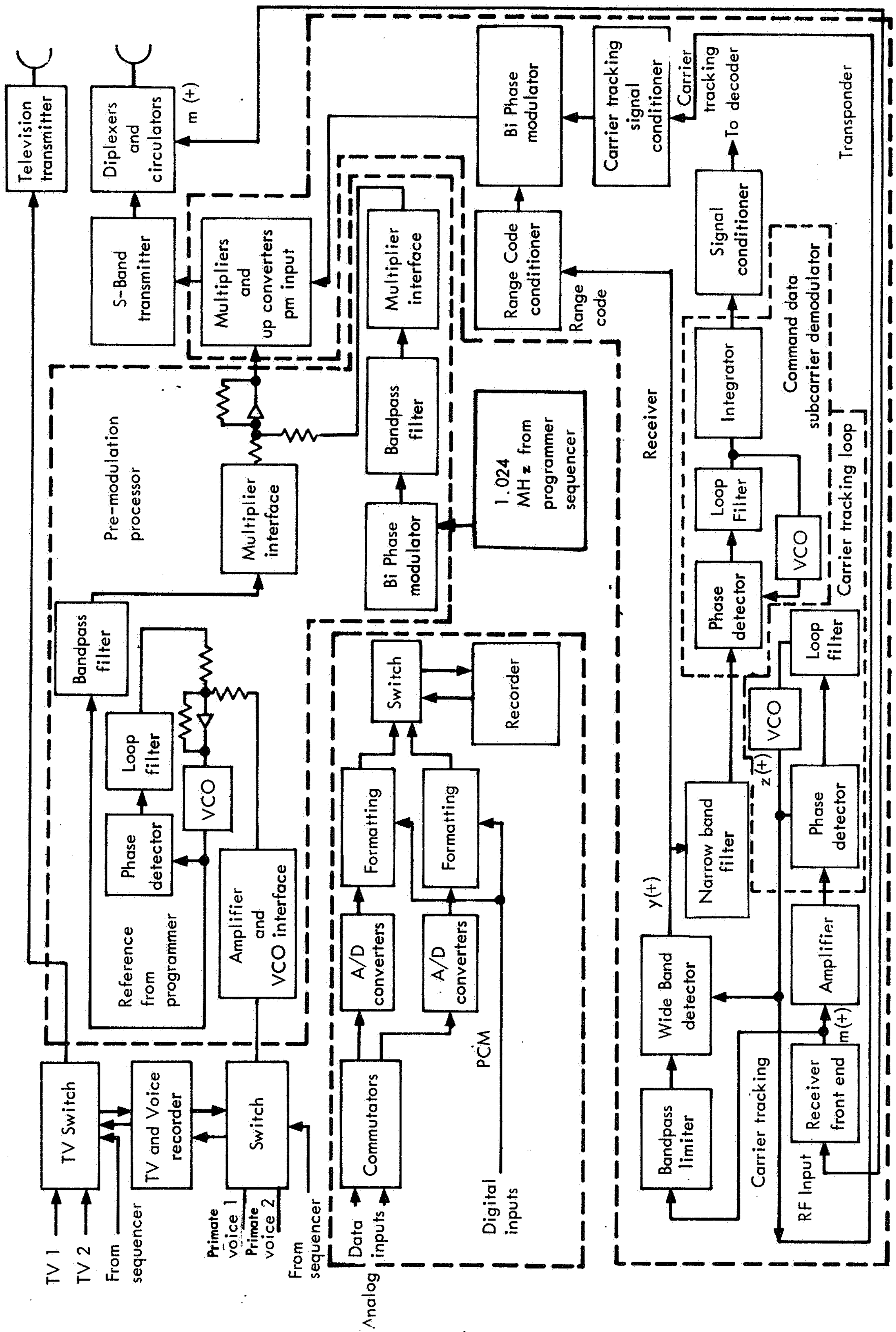


Figure 67. - Telemetry subsystem block diagram

Receiver: This section will show how the ranging code and command signals are extracted from the up-link signal.

Stored at the transmitter on the ground are two equal energy, equal duration binary waveforms $x_k(t)$, k = 1, 2, which represent the information to be sent to the spacecraft. $x_1(t)$ is the ranging information. $x_2(t)$ is the command (data) subcarrier. It is assumed that the signals $x_1(t)$ and $x_2(t)$ are normalized such that

$$E = \int_o^T x_k^2(t)dt = T$$

where T is arbitrary, and $x_k(t) = \pm 1$

The transmitted signal is a phase-modulated waveform

$$p\ (t) = \sqrt{2P}\ \sin\left[\omega_0\ t + (\cos^{-1}\ m_1)\ x_1\ (t) + (\cos^{-1}\ m_2)\ x_2\ (t)\right]$$

where ω_o is the transmitted carrier frequency, P is an arbitrary amplitude factor, and where $\cos^{-1}$ m_1 and $\cos^{-1}$ m_2 are weighting constants. Cos^{-1} m_1 is the ranging code weighting constant. Cos^{-1} m_2 is the weighting constant for the command channel.

Various factors introduce arbitrary but unknown phase shifts, θ, between the transmitted and received signals, in addition to the Doppler shift introduced by vehicle motion. The received signal on board the spacecraft is the sum of the phase shifted and frequency shifted transmitted signal and noise which is assumend to be zero mean Gaussian noise n(t) of one sided spectral density number N_o watts/Hz. The down converted received signal is given by:

$$M(t) = \sqrt{2P}\ \sin\left[\omega_1\ t + (\cos^{-1}\ m_1)\ x_1\ (t) + (\cos^{-1}\ m_2)\ x_2\ (t) + \theta\right] + n\ (t)$$

Using a trigonometric expansion, it can be shown that the above expression can be resolved into a carrier component, a ranging component, a command subcarrier component, a cross-modulation loss component and of course a noise component. In order to simplify the analysis, a further simplification will be made:

let

$$x\ (t) = x_1\ (t) + x_2\ (t)$$

then

$$v\ (t) - \sqrt{2P}\ \sin\left[\omega_1 t + (\cos^{-1}\ m)\ x\ (t) + \theta\right] + n\ (t)$$

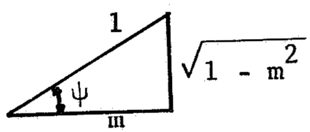
The value of the weighting constant for the ranging is no longer correct. This does not matter since the purpose of this discussion is to show how the range code and command subcarrier are derived from the input signal by means of figuring discrimination. The term v(t) can be expanded by a simple trigonometric identity into

$$v(t) = \sqrt{2P}\,\sin\left[(\omega_1 t + \theta) \cos \quad (\cos^{-1} m)\ x\ (t)\right]$$

$$+ \sqrt{2P} \cos\ (\omega_1 t + \theta) \sin \left[(\cos^{-1} m)\ x\ (t)\right] + n(t)$$

let $\psi = \cos^{-1} m$

then $\sin \psi = \sqrt{1 - m^2}$ as can be seen from the diagram below



It can be seen that $\sin \left[(\cos^{-1} m)\ x\ (t)\right]$ is an odd function. Since $x(t) = \pm 1$:

$$\sin \quad (\cos^{-1} m)\ x\ (t)\Big] = \sin\left[\ \psi\, x\ (t)\ \right] = x\ (t)\ \sin \psi$$

$$= x\ (t)\ \sqrt{1 - m^2}$$

Also $\cos \left[(\cos^{-1} m)\ x\ (t)\right] = m$ since the cosine function is an even function.

Therefore by substitution:

$$v(t) = \sqrt{2Pm^2}\,\sin\ (\omega_1 t + \theta) + \sqrt{(2P \quad (1-m^2)}\ \ x\ (t)\ \cos\ (\omega_1 t + \quad) + n(t)$$

The first term is the frequency shifted carrier component. The second term is the modulation on the carrier which consists of the range code and the command subcarrier. As can be seen from the receiver portion of the block diagram, the rf input after passing through the receiver front end and amplifier is multiplied by the VCO output in a phase detector. The output of the VCO is

$$Z(t) = \sqrt{2} \cos\ (\omega_1 t + \hat{\theta})$$

where $\hat{\theta}$ is the loop estimate of the phase of the input signal.

The wideband detector multiplies v(t) and Z(t). Assume that the double frequency components are filtered out and denote the output of the wideband detector by y(t). Then since Z(t) is

$Z(t) = \sqrt{2}\ \cos\ (\omega_1 t + \hat{\theta})$, y (t) is

$$y(t) = 2\,\sqrt{Pm^2}\,\sin\ (\omega_1 t + \theta)\ \cos\ (\omega_1 t + \hat{\theta})$$

$$+2\,\sqrt{P\ (1 - m^2)}\ x(t)\ \cos\ (\omega_1 t + \theta)\ \cos\ (\omega_1 + \hat{\theta}) + n^1\ (t)$$

where $n^1(t)$ is a Gaussian random process confined to the bandwidth of the wideband detector. y(t) can be simplified using trigonometric identities.

$$y(t) = \sqrt{Pm^2}\sin\phi + \sqrt{P(1-m^2)}\, x(t)\cos\phi + n^1(t)$$

$$\text{where } \phi = \theta - \hat{\theta}$$

Now x(t) is the modulation which consists of the normalized range code and the command subcarrier. By subsituting $x(t) = x_1(t) + x_2(t)$, y(t) can be expressed as

$$y(t) - \sqrt{Pm^2}\sin\phi + \sqrt{P(1-m^2)}\, x_1(t)\cos\phi + \sqrt{P(1-m^2)}\, x_2(t)\cos\phi + n^1(t)$$

The output of the wideband detector y(t) consists of a distortion term due to the loop error, the command data subcarrier, the range code, and a Gaussian random process respectively as shown.

Note that the distortion term is centered about zero frequency while the data subcarrier term is centered about the data subcarrier frequency of 70 KHz and occupies a bandwidth determined by the maximum deviation of the data subcarrier. The ranging code has a data rate of 1 MHz and has a power spectral density of the form $\sin^2 x/x^2$ centered about zero frequency, see figure 68.

The data subcarrier is filtered out by a bandpass filter and is routed to the command subcarrier demodulator. The ranging code will modulate the downlink signal. At the ground station the phase of the received ranging code will be compared with the phase of the transmitted ranging code. Since the velocity of propagation of electromagnetic radiation is known, the time difference between the received and transmitted codes may be translated into range. The code used is a pseudorandom code the length of which (in time) is greater than the maximum two way transmission time between the ground station and the spacecraft.

As a numerical example, to illustrate that the ranging and command codes can be extracted, suppose P (power) = 1 W for convenience, M = .2 radians and ϕ, the loop error, has an mean squared value of .1 radian (this corresponds to a signal-to-noise ratio in the loop of 10 db).

Then y (t) is given by

$$y(t) = \sqrt{.04}\sin(.316\text{ rad}) + \sqrt{.96}\, x_1(t)\cos(.316\text{ rad.})$$
$$+ .96\; x_2(t)\cos(.316\text{ rad.}) + n^1(t)$$

$$= (.2)(.311) + (.98)(.950)\, x_1(t) + (.98)(.950)\, x_2(t) + n^1(t)$$

$$y(t) = .0622 + .932\, x_1(t) + .932\, x_2(t) + n^1(t)$$

The distortion term with these conditions is then 35.6 db below either of the two desired signals, the ranging code and the command signal.

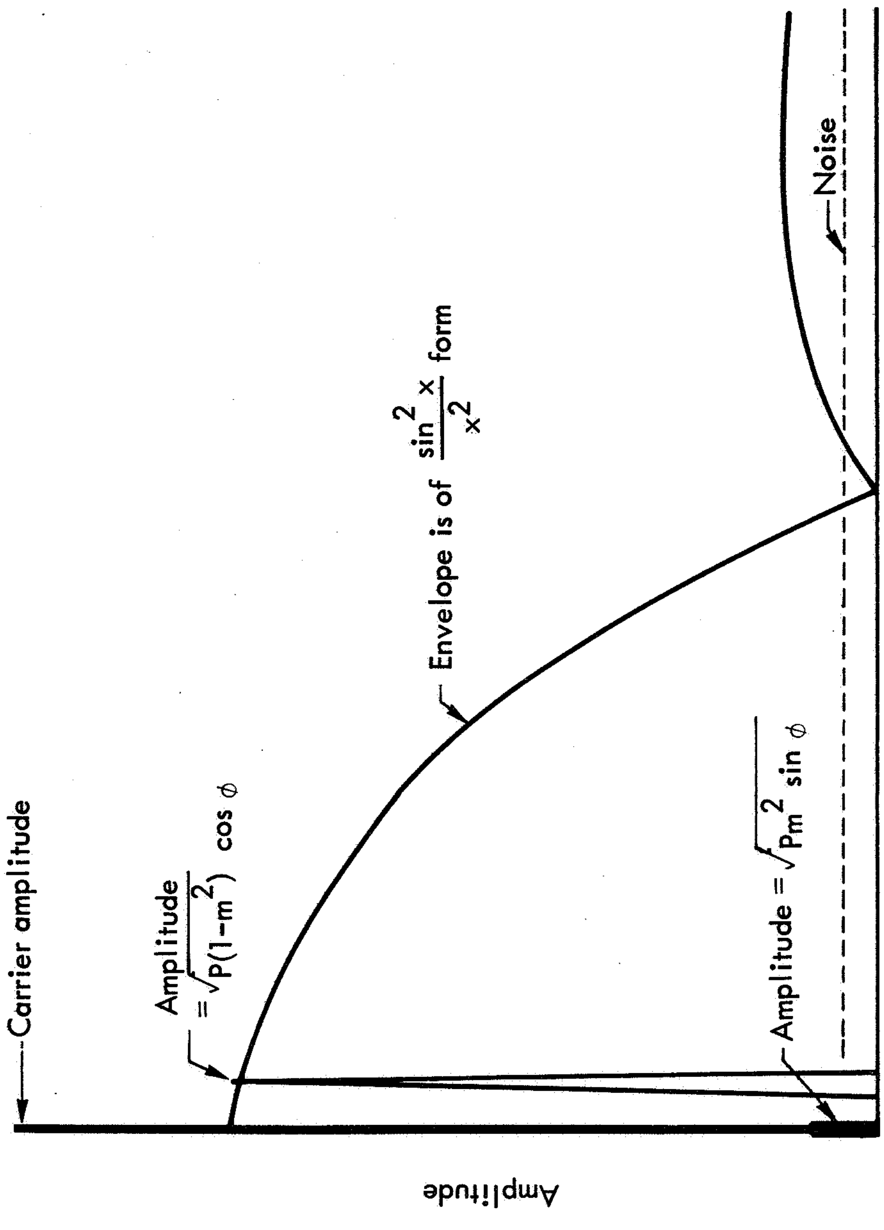


Figure 68. - Power spectral density of wideband detector output

The command data subcarrier is demodulated by a phase locked loop demodulator. The VCO in this loop is coherently related to the VCO in the carrier tracking loop. The data demodulator loop is preceded by a narrow band filter to separate out the subcarrier and to minimize the noise power input to the demodulator.

The output of the command data demodulator VCO is an estimate of the phase of the input signal. The input control signal to this VCO is an estimate of the frequency modulation of the signal since the VCO functions as an integrator in the loop. Since the input signal is a phase modulated signal, the modulation can be recovered by passing the input to the VCO through an integrator prior to signal conditioning.

Figure 69 shows a functional block diagram of the command data demodulator.

The following is a detailed analysis of the operation of the data subcarrier demodulator. The input signal is taken to be $\sqrt{2A}\,\sin\,(\omega t + \theta_1)$. The output of the VCO, which is an estimate of the input phase, is $\sqrt{2}\ K\ \cos\ (\omega t + \hat{\theta}_1)$. K is the rms output voltage of the VCO and $\hat{\theta}_1$ is the loop estimate of the input Phase θ_1. This type of loop is known as a quadrature loop because the two signals into the phase detector are in quadrature with each other. That this must be the case will be seen when the phase detector operation is explained. The phase detector multiplies the two signals as follows:

$$\sqrt{2A}\ \sin\ (\omega t + \theta_1)\cdot\sqrt{2}\ K\ \cos\ (\omega t + \theta_1) = 2\ \frac{KA}{2}\left[\sin\ (2\omega t + \theta_1 + \theta_1)\ \sin\ (\theta_1 - \theta_1)\right]$$

Now the multiplier is such that it cannot respond to the double frequency term. Therefore, the output of the phase detector is KA $\sin\ (\theta_1 - \hat{\theta}_1)$. The VCO cannot be of the form $\sqrt{2K}\ \sin\ (\omega t + \theta_1)$ because the phase detector output would then be,

$$\sqrt{2A}\ \sin\ (\omega t + \theta_1)\ \ \sqrt{2K}\ \sin\ (\omega t + \hat{\theta}_1) = AK\left[\cos\ (\theta_1 - \hat{\theta}_1) - \cos\ (2\omega + \theta_1 + \hat{\theta}_1)\right] = AK\ \cos\ (\theta_1 - \hat{\theta}_1),$$

when the double frequency terms are filtered out. In this case a good estimate of the input phase by the loop results in a maximum loop error signal, thus forcing the VCO to shift its operating point. This is clearly an impossible situation and hence, the VCO output must be of the form

$$\sqrt{2K}\ \cos\ (\omega t + \hat{\theta}_1).$$

By using Laplace transform notation the following equations are obtained:

$$V_d\ (s) = K_d\left[\theta_1\ (s) - \hat{\theta}_1\ (s)\right]$$

$$V_2\ (s) = F\ (s)\ V_d\ (s)$$

$$\hat{\theta}_1\ (s) = \frac{K_o\ V_2\ (s)}{s}$$

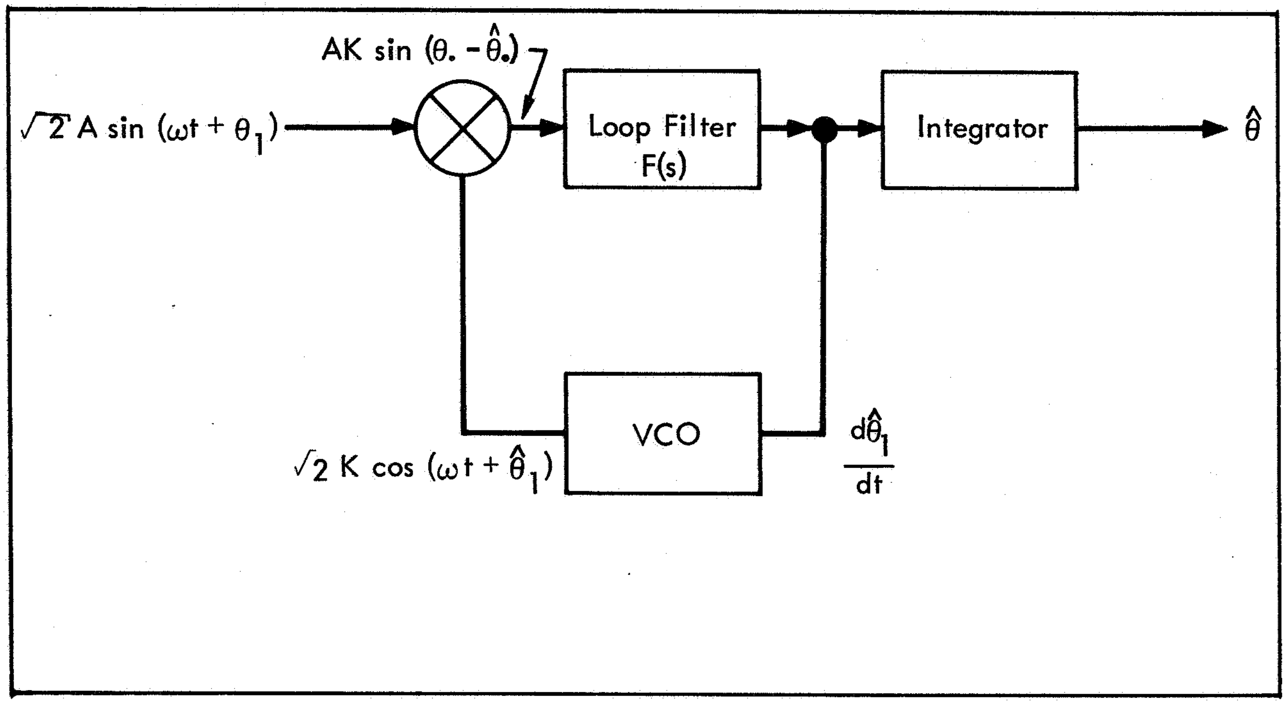


Figure 69. - Command data subcarrier demodulator

where V_d (t) = phase detector output voltage

K_d = phase detector gain constant

V_2 (t) = input voltage to VCO

F (s) = loop filter transfer function

K_o = VCO gain constant

By algebraic manipulation the transfer function of the phase error with respect to the input phase may be obtained.

$$\frac{\theta_1 (s) - \hat{\theta}_1 (s)}{\theta_1} = \frac{\theta_e}{\theta_1} = \frac{s}{s + K_o K_d F(s)}$$

The zero at the origin has acted as a differentiator resulting in the output of the loop filter being an estimate of the derivative of the input phase.

One can arrive at the same conclusion by noting that the VCO phase is an estimate of the phase of the incoming signal. However, the output frequency of the VCO is proportional to V_2, the input voltage to the VCO. That is,

$$\theta_1 = \int \omega dt = \int K_o \quad v_2 (t) \quad dt$$

Therefore, the input signal V_2 (t) is proportional to $\frac{d \hat{\theta}}{dt}$

Digital data encoding, storage, and modulation: A block diagram showing data handling for data received from the spacecraft subsystems is given in figure 70. The data points are sampled by commutators, analog-to-digital converted where necessary, and except for ECG are buffer-stored prior to transmission or recorded for subsequent transmission. The ECG samples are analog-to-digital converted, PCM formated and directly recorded.

In figure 70, analog, digital, and event data are designated by the letters A, D, and E respectively. All analog signals are converted to 8-bit digital words. Events are also coded into a standard 8-bit format. Low-rate engineering data is shown in the upper part of the diagram and the relatively high-rate primate data is shown in the lower part of the diagram.

These data and their rates are summarized in table 42. Total data bits obtained during a 24-hour day were calculated as follows:

$B = T \times R \times P \times B$

where: B = Total data bits

T = Duration of sample period, in seconds

R = Rate at which each channel is sampled during the sample period, in samples per second

P = Total number of sample periods per day for all channels

B = Bits required for each sample

= 8 bits for all data except Behavioral Panel, Waterer & Feeder (refer to table 42 for details)

The total bits summarized in table 42 must be dumped downlink each day. However, the maximum spacecraft record-time capability required is approximately 16 hours since the maximum number of orbits not passing over a given ground station is ten. During a 16 hour period, approximately two-thirds of the data in table 42 will be acquired; therefore, the total on-board recording capability required is approximately 16×10^6 bits.

The recording approach will be based on the ECG data because of the relatively high bit rate and total storage requirements. The scheduled approach will be to start the recorder every four hours, dump buffer-storage, then sample and record each ECG channel. Otherwise, when buffer storage is filled, the buffer storage is unloaded into the tape recorder.

A four-track recorder running at 3.75 inches per second (ips) is capable of recording 6.4 Kbps. This is the rate required to record an ECG channel. A play-back-to-record ratio of 8:1 provides the standard Apollo dump rate of 51.2 Kbps. To record 16×10^6 bits of information at 6.4 Kbps requires 2,500 seconds, or approximately 780 feet of four-track for the 3.75 ips tape speed.

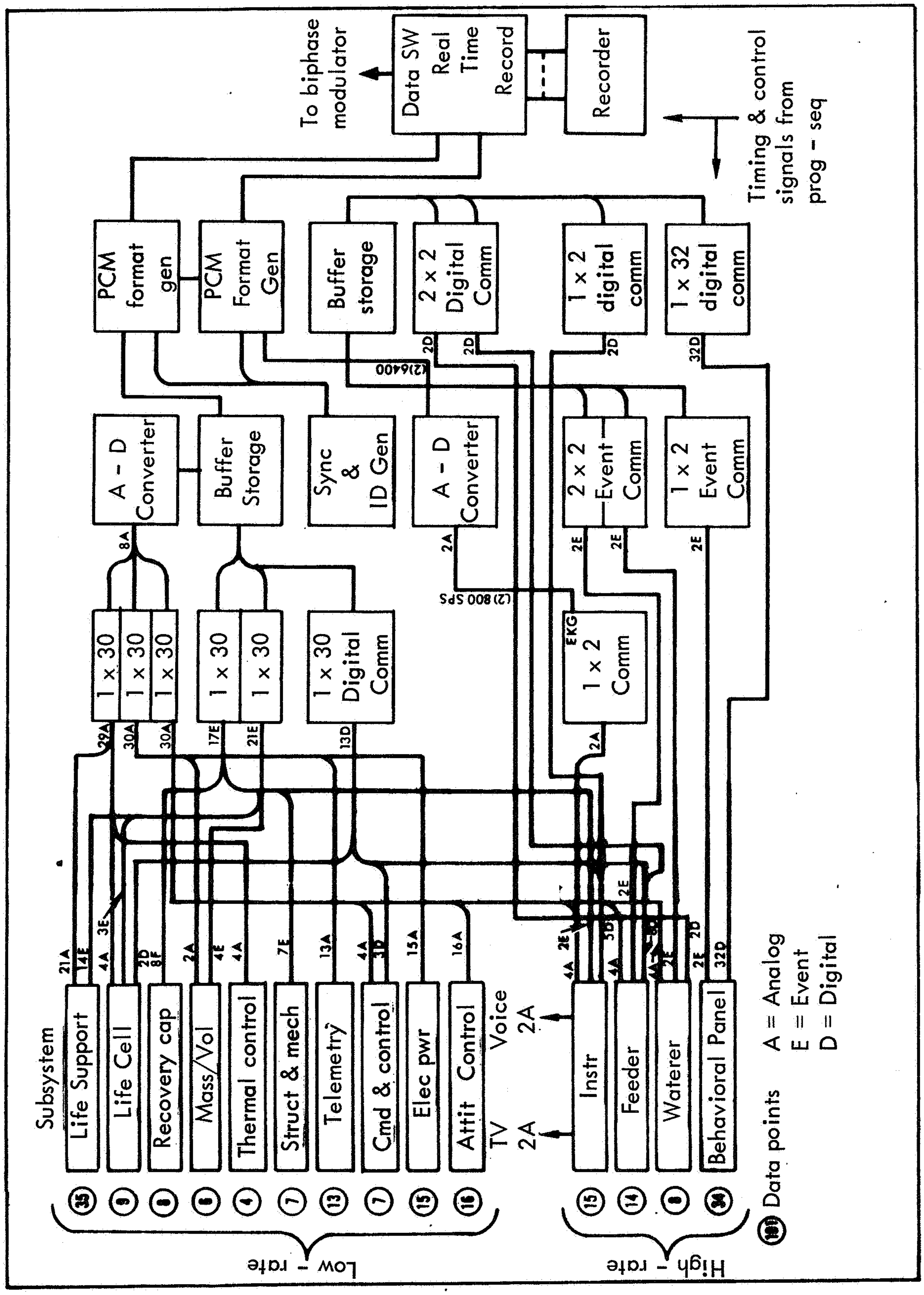


Figure 70. - Data handling block diagram

TABLE 42.- SUMMARY OF DAILY DATA REQUIREMENTS

No. of channels	Parameter	Signal bandwidth	Sample period duration	Sampling rate (SPS)	Total sample period per 24-hr day	Total data per 24-hr day (K-bits)
(2)	EKG	0 - 100 Hz	300 Sec.	800	12	23,040.0
(2)	Primate Temp.	0 - 0.001 Hz	1 Sec.	1	576	4.6
(2)	Respiration Rate	20 - 60 BPM	1 Sec.	1	576	4.6
(42)	Behavioral Panel Waterer, and Feeder	--- ---	Continous	As Occurs	28,620	252 .2
(72)	Misc. Life Support	0 - 0.001 Hz	1 Sec.	1	2,283	18.3
(4)	Thermal Control	0 - 0.001 Hz	1 Sec.	1	16	0.1
(7)	Struct. & Mech.	0 - 0.001 Hz	1 Sec.	1	7	0.1
(5)	Instrumentation	0 - 0.001 Hz	1 Sec.	1	2,889	23.1
(13)	Telemetry	0 - 0.001 Hz	1 Sec.	1	52	0.5
(7)	Command & Control	0 - 0.001 Hz	1 Sec.	1	28	0.2
(15)	Electrical Power	0 - 0.001 Hz	1 Sec.	1	222	1.8
(16)	Attitude Control	0 - 0.001 Hz	1 Sec.	1	159	1.3
					TOTAL	23,346.8

The time required to dump 24 x 10^6 bits downlink at a 51.2 Kbps rate is approximately 8 minutes. The minimum expected over-station transmission time is 30.1 minutes per day, which is more than adequate to dump all data.

As shown in figure 71, data will ordinarily be recorded in the following sequence every four hours: buffer storage data followed by samples of ECG. The recorder will be started at any time the buffer-storage is filled to prevent loss of data due to overflow.

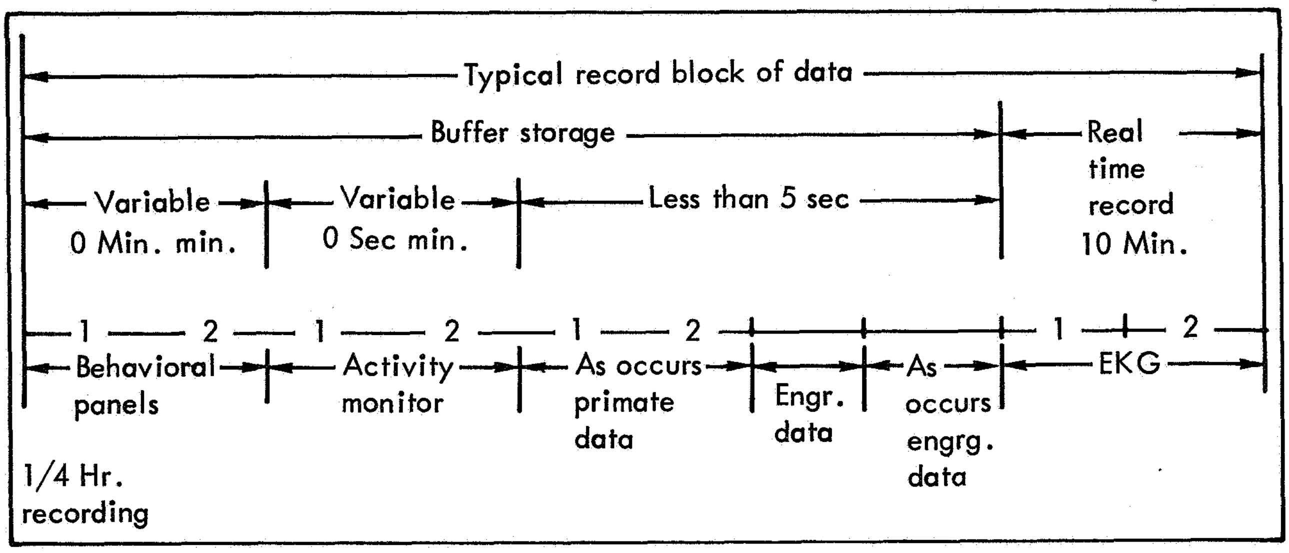


Figure 71. - Data recording format

The tape recorder will record 4 tracks at a data rate of 1.6 Kbps per track. This requires a tape speed of 3.75 inches per second. The playback rate is 51.2 Kbps which corresponds to a tape speed of 30 inches per second. A 2200 foot tape has a capacity of 426 bits per inch per track or 5112 bits per foot per track (for 1.6 Kbps at 3.75 cps). The total capacity for four tracks of a 2200 foot tape is approximately 50 x 10^6 bits (5112 bits per foot x 2200 feet x 4 tracks). This is sufficient since approximately 16 x 10^6 bits of recorder capability are required for 10 orbits.

Recording of the data onto tape is 4-track parallel at a 6.4 Kbps rate. The 4 most significant bits of the 8 bits are read-on in parallel followed by parallel recording of the 4 least significant bits. This gives a 1.6 Kbps recording rate on each track at 3.75 ips. During playback of 30 i.p.s. (ratio 8:1) the data is read-off in parallel at 12.8 Kbps and regrouped to a serial stream of 51.2 Kbps in preparation for down-link transmission. The down-link format is compatible with the Apollo facilities, 128 words (8 bit words) per frame at 50 frames per second. The frame length is fixed by incorporating zeros to supplement partial frames of data when necessary.

Several areas must be considered in arriving at a realistic record cycle. A minimum number of record cycles would introduce an unreasonable size of buffer storage while a maximum number of record cycles will require a large number of start and stop recorder operations during the year mission. For example, a record cycle every 5 minutes will require over 10^5 start-stops per year.

In addition to the failure caused by starts and stops, tape wear is also a factor to consider. However, for the Primate Spacecraft tape wear is not expected to be a problem. Generally speaking, playback will occur once per day or 365 times per year, and the maximum number would be every orbit over the ground station which would be near 6 per day or approximately 2,000 per year. This figure is not unreasonable for 1 mil tape, however, difficulty has been experienced with 0.5 mil tape at passes greater than 2000 per year. 50,000 start-stops of recorders have been achieved by incorporating solid-state switching. Using 20,000 as a figure for desirable start-stops dictates a buffer storage of approximately 6000 bits as shown below. A storage of this size would limit start-stops of the recorder to approximately 55 per day.

The behavioral panel, waterer, and feeder data quantities in table 42 were based on: A maximum handle activation rate of 10 per second for the ILK, TIM, and VIG toggles, and the need to record the number of handle activations and the time of day at which the combined reward lamp comes on, to the nearest 0.01 sec with respect to the time at which the stimulus lamp comes on.

Behavioral panel, waterer, and feeder data are given in more detail in table 43. Since, unlike other data, these events occur at random times and intervals, a buffer-storage is required to minimize recorder start-stop cycles and wear, and to make efficient utilization of recorder tape capacity. Since ECG is recorded when sampled, and hence, requires no buffer storage, the remaining 327×10^3 bits of data in table 42 must be buffer-stored every 24 hours.

The data recorder is run for five minutes six times per day to record ECG and six times per day to dump data. If the recorder is not to be turned on except when necessary to record ECG or dump data, the minimum buffer-storage required for all spacecraft data would be 327×10^3 bits/day $\div$ 12 operations/day $= 28 \times 10^3$ bits/operation where, "operation" means a recorder start-stop cycle which coincides with buffer storage of the data. However, since as many as 20,000 per year (55 per day) recorder start-stop operations can be reliably achieved, an adequate total buffer storage size is:

$$\frac{327 \times 10^3 \text{ bits/day}}{55 \text{ operations/day}} = 6000 \text{ bits}$$

Since non-interrupted recording of ECG data is desirable, and the maximum number of handle actuations in any five-minute period is 300 (per primate), the maximum Behavioral Panel buffer-storage required would be:

$$2 \text{ primates} \times 300 \text{ events/primate} \times 6 \text{ bits/event} = 3600 \text{ bits}$$

TABLE 43. - BEHAVIORAL PANEL, WATERER, AND FEEDER TYPICAL OPERATION SEQUENCE

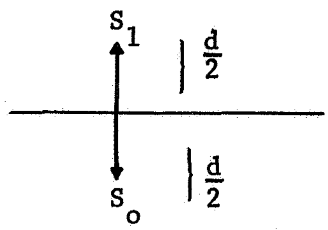
Total events per day	Total bits per day	Event	Action
		(1) Handle lamp on	Record time of day (BCD) to nearest 0.1 second (24 bits)
236	1,888	a) ILK	
236	1,888	b) TIM	
236	1,888	c) VIG	
8	64	d) EXC	
		(2) Handle actuated *	Record elapsed time between successive handle actuations (6 bits) and identify handle (2 bits)
23,600	188,800	a) ILK	
236	1,888	b)TIM, or	
236	1,888	c) VIG	
8	64	d) EXC	
716	4,296	(3) Reward Lamp on	Record elapsed time with respect to (1) above (6 bits)
		(4) Mouthpiece switch activated	Record elapsed time with respect to (1) above (6 bits) and identify switch (1 bit)
300	2,100	a) Food	
416	2,912	b) Water	
716	4,296	(5) Reward Delivery	Record elapsed time with respect to (1) above (6 bits)
1,676	40,324	(6) Miscellaneous	Record time of day (BCD) to nearest 0.1 seconds (24 bits)
28,620	252,196	TOTALS	

* TIM and VIG activated 118 times per day each, ILK is activated 11,800 times per/day.

Hence, using the 6000 bit buffer-storage will assure uninterrupted recording of ECG data and will limit recorder start-stop operations to known capabilities.

Proposed time formats for the behavioral panel, waterer and feeder data are given in figures 72 and 73.

In order to be compatible with the MSFN, the data bi-phase modulates a 1.024 MHz sine wave produced by a crystal oscillator. The signals produced are the optimum binary signals and are termed antipodal. That is, the two states 0 and 1 for the digital data correspond to phase shifts of the sine wave of + 90° and -90°. These two signals are represented by a vector diagram below.



Where S_o represents a "0" and S_1 represents a "1". The horizontal line through the origin represents the boundary between the decision regions for S_o and S_1. Since a "0" is mistaken for a 1 and vice versa, if and only if the noise exceeds d/2 (which is the distance from the tip of the signal vector to the boundary of the decision region) it is clear that an angular difference of 180° minimizes the probability of the noise vector exceeding the signal vector. This therefore, shows the optimality of antipodal binary signals as opposed to orthogonal signals for example. (Orthogonal signals are represented by two vectors at right angles to each other.)

The probability of error given S_1 is the probability the noise vector n exceeds d/2. But n is zero - mean Gaussian with variance (or two sided noise spectral density No/2) so that

$$P\left[e/s_1\right] = \int_{d/2}^{\infty} \frac{1}{\sqrt{\pi N_o}} e^{-\frac{x^2}{\pi N_o}} dx$$

The overall probability of error is

$$P(e) = \sum_{i=o}^{1} P\left[S_i\right] P\left[e/S_i\right] = P\left[e/S_o\right] = P\left[e/S_1\right]$$

since S_o and S_1 are equiprobable. When the signals have minimum energy, the length of each vector is $\sqrt{E_s}$ where E_s is the signal power, then $d/2 = \sqrt{E_s}$.

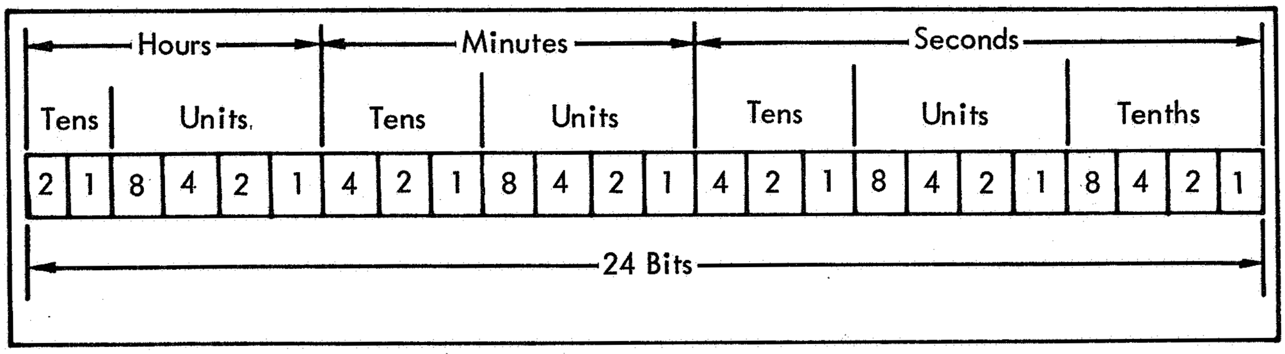
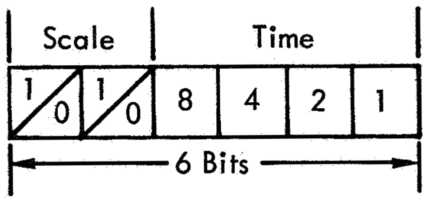


Figure 72. - One tenth second time format



Scale	Time
00	Tens of millisec
01	Tenths of sec
10	Units of sec
11	Tens of sec

Figure 73. - Elapsed time format

Define $\gamma = X\sqrt{2/N_o}$. Then the probability of error is given by

$$P(e) = \int_{\sqrt{E_s}/\sqrt{N_o/2}}^{\infty} \frac{1}{\sqrt{2\pi}} e^{-\frac{\gamma^2}{2}} d\gamma = \text{erfc} \sqrt{2E_s/N_o}$$

where erfc x = 1 - erf x and erfx is the error function defined by

$$\text{erf}x = \frac{1}{\sqrt{2\pi}} \int_o^x e^{-y^2/2} dy$$

Primate video and voice: The following section summarizes the requirements which must be met by the video recorder in order to record Apollo quality TV during the time when the spacecraft is not over a ground station.

Figure 74 shows a time line of TV recording for one day.

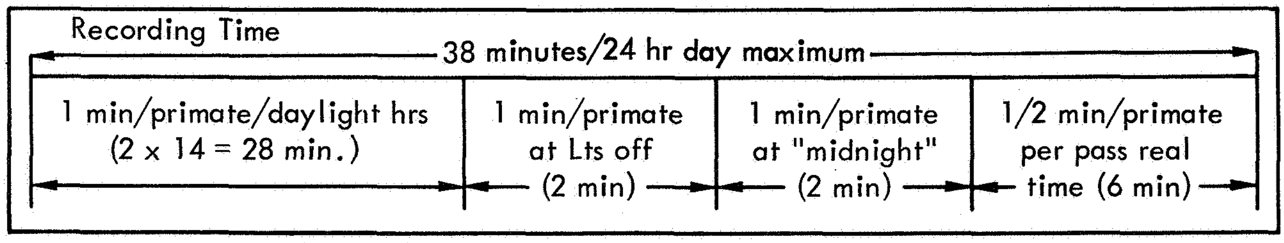


Figure 74. - Video time line recording for one day

A recording time of 32 minutes maximum per 24 hour day (10 orbits not over station) plus start and stop time for 365 days yields a recording time of 10,950 minutes (plus start and stop time) for the complete mission.

Recorder Playback (dump) is done at a 1:1 playback to record ratio. A minimum of 30.1 minutes will be available for playback in a 24-hour period. Therefore, not all of the recorded video will be replayed during some portions of the year. During other portions of the year sufficient time will be available for complete playback. The over station time for a 24-hour period varies from 44.6 minutes maximum to 40.1 minutes minimum. Allowing one minute acquisition time per pass plus one minute real time per pass allows only 30.1 minutes for playback on a "5 pass" day and 32.6 minutes for a "6 pass" day. The television time line is shown in figure 75.

The video tape recorder is required to have a peak-to-peak signal to rms noise ratio of 25 db minimum in conformance with the output of the television camera.

Recorder bandwidth will be suitable for recording a camera output having the following characteristics: ± 3 db from 40 Hz to 400 KHz and +3 to -12 db from 400 KHz to 500 KHz. Roll off above 500 KHz will not be less than 20 db per octave.

A voice track will be included in the video tape recorder. The response of this recorder will be (as a minimum) from 50 Hz to 12 KHz. Primate voice will be recorded only at the same time as primate video.

The primate voice must frequency modulate a subcarrier of 1.25 MHz. In order to remove spurious FM, due to oscillator instabilities, the 1.25 MHz frequency is obtained from the master oscillator in the programmer. This is possible since 1.25 MHz is an integral multiple of the data rate 51.2Kbps. The reference from the programmer controls the frequency of a VCO in a phase locked loop. The modulation is introduced by varying the control voltage to the VCO.

Transmitters: In this section the link analysis for the television link (FM) for the Primate Spacecraft is discussed. Two link analyses have been performed. Since the demodulator at the MSFN station has a loop equivalent bandwidth which is selectable, i.e., 4 MHz or 11 MHz. The loop equivalent noise bandwidth, β_n is related to the loop natural frequency ω_n by the following expression:

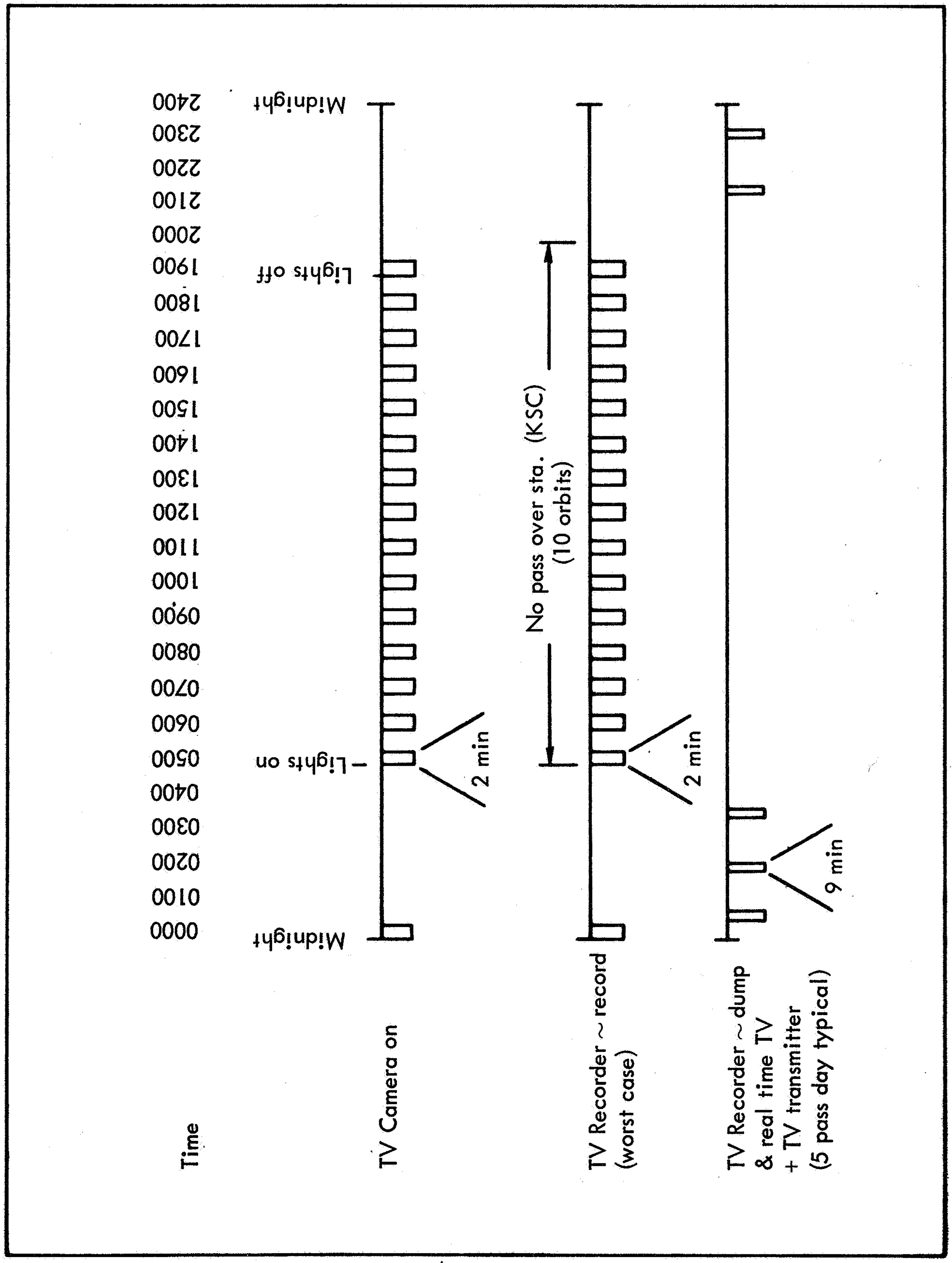


Figure 75. - TV time line

$$\beta_n = \frac{\omega_n}{2\zeta} (1 + 4\zeta^2)$$

where ζ is the damping factor. For optimum loop performance, $\zeta = .707$. Then ω_n is the 3 db point of the closed loop frequency response. Thus, ω_n is the information bandwidth of the loop. When = .707, the expression reduces to

$$\beta_n = 6.73\ \omega_n$$

The 3 db of the output of the television camera is 400 KHz. Thus, the information bandwidth of the television signal is 400 KHz. Since the receiver bandwidth is fixed, either 4 MHz or 11 MHz equivalent noise bandwidth, the maximum carrier deviation is determined by having the sum of the information bandwidth of the transmitted signal and the carrier deviation equal to the receiver information bandwidth. The deviations and receiver information bandwidths are summarized in the table below:

TABLE 44. - RECEIVER INFORMATION BANDWIDTHS AND DEVIATIONS

Receiver equivalent noise bandwidth	Receiver information bandwidth	Carrier deviation
4 MHz	600 KHz	200 KHz
11 MHz	1.65 MHz	1.25 MHz

Since the carrier deviation for both cases has been determined, it is necessary to determine the modulation index. In order to calculate the modulation index, it is necessary to determine the average modulating frequency. As a first approximation, it is assumed that the highest frequency out of the television camera is 400 KHz and that this corresponds to a picture such that alternating picture elements are black and white. The camera output has a minimum of five shades of gray. If it is assumed that an average picture consists of smooth transitions of black through five shades of gray to white such that the average frequency out of the television camera is 400 KHz ÷ 5 = 80 KHz, then the average modulation index is:

$$\frac{200}{80} = 2.5$$

for the 4 MHz loop equivalent noise bandwidth case and for the 11 MHz loop equivalent noise bandwidth case.

$$\frac{1250}{80} = 15.6$$

It is possible to decrease the mean square modulating frequency, and hence increase the modulation index, by adding a pre-emphasis network on the spacecraft which emphasizes the lower frequencies. This is compensated on the ground by incorporating a de-emphasis network in the ground demodulator. Examination of the first link analysis shows an output SNR at 5° above the horizon of 22.1 db and a margin of 5.1 db. The second link analysis, for the wide band case, has an output SNR of 39.8 db but only a margin of 1.1 db. The significance of the low margin is that the difference between the input SNR into the loop and the SNR required for a 99.8%, 3σ, probability of staying in lock is equal to the margin. For the wide band case, while the loop is in lock, the output SNR is 39.8 db. However, a small decrease in input signal could cause the loop to lose lock resulting in a loss of data. Therefore, the best choice is to use the narrow band signal at least for initial acquisition. It would be possible to shift to the wide band mode when the spacecraft was closer.

The FM improvement factor is normally given as $3\beta^2$. However, this expression is not strictly correct for large values of β. Therefore, in order to be safe in calculating the FM improvement factor for the wideband case, a value of 10 for β was used instead of the actual modulation index 15.6.

The link analysis assumed the following pertinent parameters:

(1) 20 W transmitter

(2) 1300 mile maximum slant range

(3) An antenna having a beam-width of 90° and a maximum gain of 9 db

(4) A cooled parametric amplifier with the antenna pointing at the horizon, worst case, yielding a system noise temperature of 220°K

Results of link analyses for wide and narrow band television are given in tables 45 and 46.

The voice is frequency modulated onto a subcarrier of 1.25 MHz while the telemetry data is bi-phase modulated onto a 1.024 MHz subcarrier. The two subcarriers, corresponding to primate voice and data then are summed in an operational amplifier and are used to phase modulate the transmitted signal. The input to the transmitter is the carrier tracking signal generated by the receiver which has been multiplied, up converted and phase modulated by the voice and data subcarriers to produce the output signal.

The analysis assumes the following pertinent parameters:

(1) Five watt transmitter

(2) Maximum slant range of 1300 miles

(3) An omnidirectional antenna having a maximum gain of +4.5 db and pattern nulls having a gain of -10 db

(4) Cooled parametric amplifier with the antenna pointing at the horizon (worst case) yielding a system noise temperature of 220°K.

TABLE 45. - TELECOMMUNICATION DESIGN CONTROL
FM, NARROW BAND TELEVISION

Parameter	Value	Tolerance
Total transmitter power	43 dbm	+ 1.0 - 1.0
Transmitting circuit loss	- 2 db	+ .4 - .4
Transmitting antenna gain	9 db	
Transmitting antenna pointing loss	- 3 db	
Space loss	-167.3 db	+ 1.0 - 1.0
@ 2272.5 MC, R = 1300 miles		
Polarization loss	- 3 db	+3.0 - 0.0
Receiving antenna gain	+ 43.9 db	+ .5 - .5
Receiving antenna pointing loss	----	
Receiving circuit loss	.2 db	+.1 - +.1
Net circuit loss	-175.3 db	+5.0 - 3.0
Total received power	-99.6 dbm	+ 6.0 - 4.0
Receiver noise spectral density (N/B)	-175.2 dbm	+ 0.0 - 1.5
T system = 220°K		
Carrier APC noise BW ($2B_{LO}$= 4MHz)	66 db	
FM improvement factor $3\beta^2$ β =2.5	12.7 db	
Noise power	109.2 dbm	+0.0 - 1.5
Performance		
Threshold SNR in $2B_{LO}$	4.5 DB	+ 1.0 -1.0
Threshold carrier power	-104.7 dbm	-1.5
Performance margin	+5.1 db	+7.0 - 8.5
Output SNR	22.1 db	+7.0 - 8.5

TABLE 46. - TELECOMMUNICATION DESIGN CONTROL.

FM, WIDE BAND TELEVISION

Parameter	Value	Tolerance
Total transmitter power	43 dbm	+1.0 - 1.0
Transmitting circuit loss	- 2 db	+ .4 - .4
Transmitting antenna gain	9 db	
Transmitting antenna pointing loss	- 3 db	
Space loss	-167.3 db	+1.0 - 1.0
@ 2272.5 MC, R = 1300 miles		
Polarization loss	-3 db	+3.0 - 0.0
Receiving antenna gain	+ 43.9 db	+ .5 - .5
Receiving antenna pointing loss		
Receiving circuit loss	.2 db	+ .1 - .1
Net circuit loss	-175.6 db	+5.0 - 3.0
Total received power	-99.6 dbm	+6.0 - 4.0
Receiver noise spectral density (N/B)	-175.2 dbm	+ 0.0 - 1.5
T system = 220°K		
Carrier APC noise BW ($2B_{LO}$= 11 MHz)	70 db	
FM improvement factor $3\beta^2$ β=15.6	34 db	
Noise power	-105.2 dbm	- 1.5
Performance		
Threshold SNR in $2B_{LO}$	4.5 db	+ 1.0 - 1.0
Threshold carrier power	-100.7 dbm	- 1.5
Performance margin	+1.1 db	+7.0 - 8.5
Output SNR	39.8 db	+7.0 - 8.5

(5) A new plug-in voice amplifier is provided for primate voice to replace the present voice amplifier demodulator which has a frequency response of 50 Hz to 3.4 KHz. A conventional ratio detector type of discriminator and a phase locked loop demodulator are analyzed. The information bandwidth of the signal, and hence of the demodulator, is related to the required loop equivalent noise bandwidth for the phase locked loop demodulator. The loop equivalent noise bandwidth β_n is related to the loop natural frequency by the expression

$$\beta_n = \frac{\pi \omega_n}{2} (1 + 4\zeta^2)$$

where ζ is the damping factor. For optimum performance ζ = .707. Then ω_n is the natural frequency of the loop which is the 3 db point of the closed loop frequency response (when ζ = .707). Thus ω_n is the information bandwidth of the loop

when ζ = .707, the expression reduces to

$$\beta_n = 6.73\omega_n$$

If $\frac{\omega_n}{2\pi}$ = 20 KHz, then $\beta_n \cong$ 133 KHz. Primate voice is assumed to have a bandwidth of 20 KHz. A conventional discriminator may be preceded by a very sharp cut-off filter such that the noise bandwidth can be restricted to 60 KHz or less.

The modulation indices derived are as follows. The primate voice, which is assumed to have a bandwidth of 20,000 Hz, frequency modulates the 1.25 MHz subcarrier. The deviation is set at 40 KHz which results in a modulation index of 2 at 20 KHz. The data bi-phase modulates the data subcarrier.

A $P_{subcarrier}/P_{total}$ ratio of -9 db requires a peak modulation index of 0.9 radian. This applies to both the voice and data subcarriers. Results of link analysis for the data channel are given in table 47.

TABLE 47. - TELECOMMUNICATION DESIGN CONTROL
PM, CARRIER TRACKING AND VOICE DATA

Parameter	Value	Tolerance	
Total transmitter power	+ 37 dbm	+10	-1.0
Transmitting circuit loss	-2 db	+.4	-.4
Transmitting antenna gain	-10 db	+14.5	-0.0
Space loss	-167.3 db	+1.0	-1.0
@2287.5 MC, R=1300 miles			
Polarization loss	-3 db	+3.0	-0.0

TABLE 47. - continued

Parameter	Value	Tolerance	
Receiving antenna gain	+43.9 db	+.5	-.5
Receiving antenna pointing loss			
Receiving circuit loss	-.2 db	+.1	-.1
Net circuit loss	-182.5 db	+20.5	-3.0
Total received power	-101.6 dbm	+21.3	-4.0
Receiver noise spectral density (N/B)	-175.2 dbm	+0.0	-1.5
T system = 220°K			
Carrier modulation loss	-6 db	+1.0	-1.0
Received carrier power	-107.6 dbm	+22.3	-5.0
Carrier APC noise BW ($2B_{LO}$ = 700 Hz)	28.5 db		
(Selectable 700, 200, 50 Hz)			
CARRIER PERFORMANCE - TRACKING			
Threshold SNR in $2B_{LO}$	4.5 db	+1.0	-1.0
Threshold carrier power	-142.2 dbm		-1.5
Performance margin	+34.6 db	+36.7	-7.5
DATA CHANNEL			
Modulation loss	-9 db	+1.0	-1.0
Received data subcarrier power	-110.4 dbm	+21.3	-3.0
Bit rate (1/T) 51.2K bps	47.1 db	---	
Required ST/N/B	6 db	+1.0	-1.0
Threshold subcarrier power	-122.1 dbm	+1.0	-1.0
Performance margin (modulation index = .9 rad peak)	+11.7 db	+24.3	-6.0
VOICE CHANNEL (conventional discriminator)			

TABLE 47. - concluded

Parameter	Value	Tolerance	
Modulation loss	-9 db	+1.0	-1.0
Receiver voice subcarrier power	-110.4 dbm	+21.3	-3.0
Voice noise bandwidth = 60 KHz	47.8 db		
Threshold SNR in 2 B_{LO}	10 db	+1.0	-1.0
Threshold subcarrier power	-115.4 dbm	+1.0	-1.0
Performance margin (modulation index = .9 rad peak)	+6.8	+24.3	-6.0
VOICE CHANNEL (phase locked loop)			
Modulation loss	-9 db	+1.0	-1.0
Received voice subcarrier power	-110.6 dbm	+21.3	-3.0
Loop noise bandwidth ($2B_{LO}$ = 133.3 KHz)	51.3 db		
Threshold SNR in $2B_{LO}$	6.0 db	+1.0	-1.0
Threshold subcarrier power	-117.9 dbm	+1.0	-1.0
Performance margin	7.3 db	+24.3	-6.0

Antennas: The purpose of this section is to describe the gains and configurations required for the television and data antenna systems. The antenna gains required are determined by the results of the link analysis performed for the data/voice and television channels. The link analysis for the data and voice link shows that an omnidirectional antenna with pattern nulls having gains of -10 db is adequate for that link. The link analysis for the television link showed that a minimum gain of 6 db, including a 3 db pointing loss, is required for a SNR of 22.1 db. This gain, neglecting antenna losses, may be achieved by the use of a directional antenna having an on-axis gain of 9 db and beamwidth of 90°.

The orientation of the perpendicular to a fixed point on the spacecraft with respect to the ground station is random in the sense that any angular relationship is possible. This is because the spacecraft is sun-oriented and has a period of revolution about the earth of 90 minutes while the period of the earth's rotation is, of course, 24 hours. A consequence of this is that omnidirectional coverage must be achieved with both the data and television links. A relatively simple mechanization is possible for the data link since the link analysis has shown that omnidirectional antennas will provide an adequate SNR. This is not the case with the television antennas since the link analysis showed that a directive antenna is required.

The discussion to follow will be separated into two parts. The first part will discuss antenna configurations as dictated by link analysis considerations. The second part of the discussion will be concerned with the suitability of various types of antennas for the data links considering vehicle geometry and trajectory.

As was stated before, omnidirectional antennas may be used for the data/voice link. Figures 76A and 76B show the two alternate approaches for achieving omnidirectional coverage with the corresponding antenna patterns. In both cases, the antennas would have to be deployed due to the mounting configuration in the launch vehicle. Figure 76A shows the antennas located on a line colinear with the spacecraft. The dotted line in figure 76A shows the effect of possible cancellations resulting from reflections from the solar panels. It is clear that the better approach is the approach of figure 76B since it provides greater omnidirectivity and a more uniform pattern.

Since there are two antennas, command data from the ground will, in general, be received on both antennas. The antenna decision circuitry, figure 77, will determine at which antenna and receiver front end the stronger signal is being received. An inhibit signal is developed which inhibits the receiver which has the weaker output. Threshold circuitry is included so that neither channel will be inhibited in the absence of a signal. This is done to enable initial acquisition of the up-link signal.

The antenna selected for receiving the command data is used for transmitting downlink data.

The types of antennas which can be used will now be discussed. The data/voice antenna must be omnidirectional antennas giving hemispherical coverage. Two types of antennas which are suitable for this application are turnstile antennas and dielectrically loaded open-ended waveguides. The turnstile antenna consists of two one-half wavelength dipoles mounted at right angles to each other on top of a mast of suitable height. A circular ground plane is mounted below the crossed dipoles to spread out the pattern. The lengths of the cross members are trimmed during fabrication to eliminate ellipticity in the pattern and to establish circular polarization.

The dielectrically loaded open ended waveguide consists of a circular waveguide in which a circular mode propagation has been established. At the open end of this waveguide is located a dielectric lens, the function of which is to spread out the beam pattern.

Either of these two antennas would have to be deployed since it is required that the antennas be located sufficiently far from the body of the spacecraft to achieve a hemispherical cardioid pattern. The turnstile type antenna is a simpler and less expensive antenna to make and therefore, represents a better choice for the data/voice antennas.

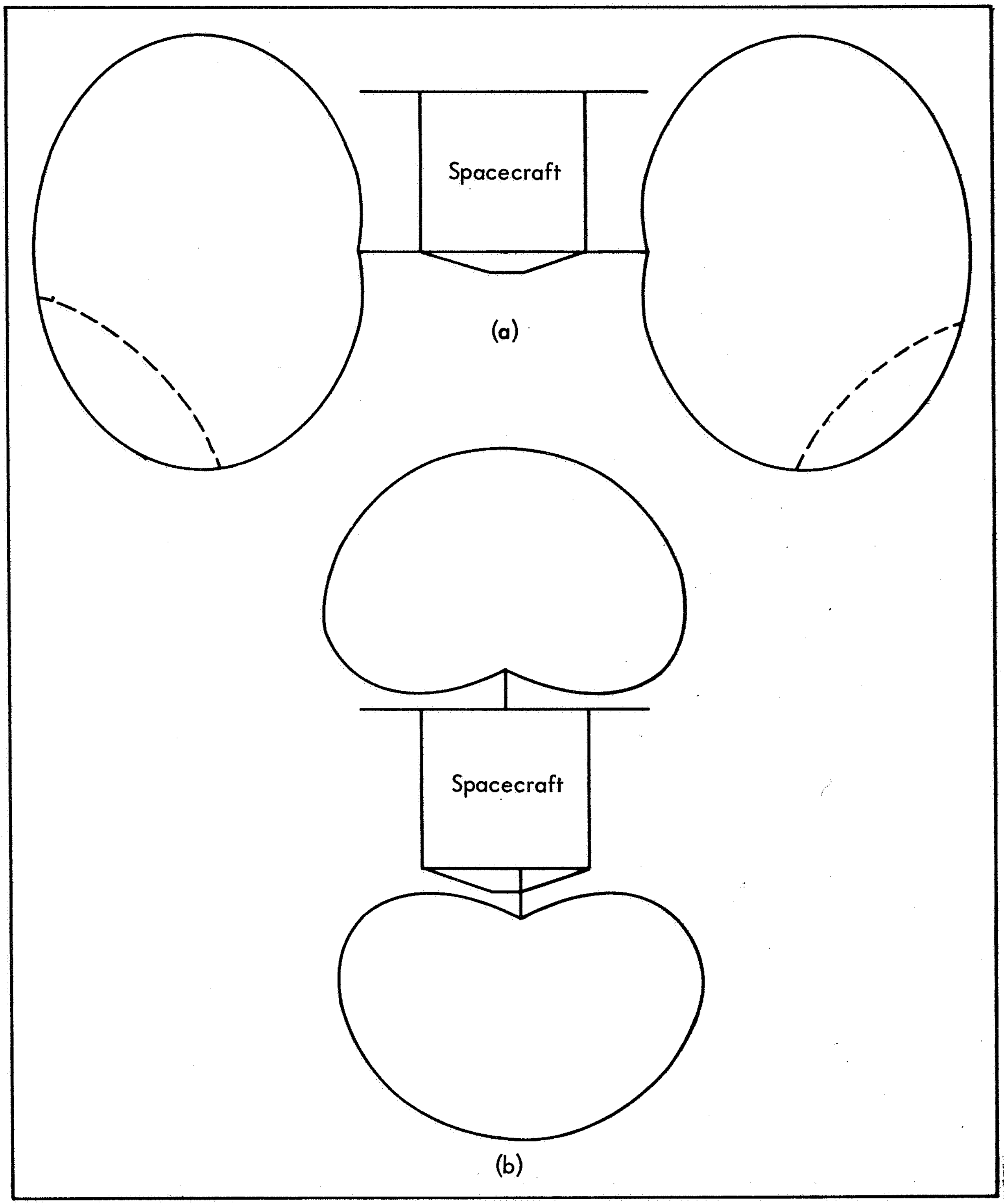


Figure 76. - Omnidirectional antenna patterns

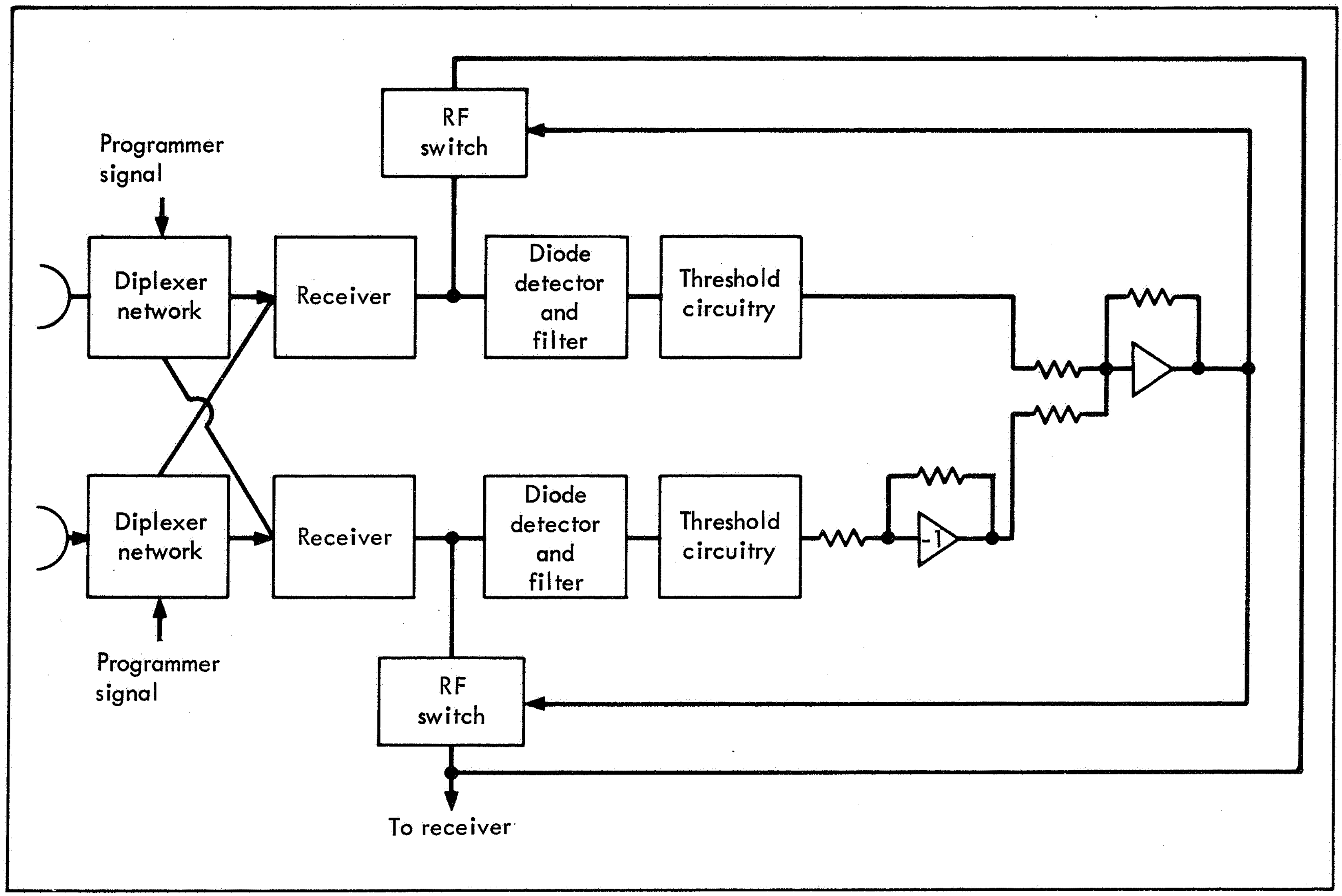


Figure 77. - Data antenna decision block diagram

The antenna configuration required for the television transmission will be discussed in this section. The link analysis has shown that a minimum antenna gain of 6 db (with a transmitter power of 20W) is needed for television transmission. Clearly, this cannot be done with omnidirectional antennas. Furthermore, since the full transmitter power is required, some means must be provided for selecting one out of several antennas.

An antenna which has a peak gain of 9db will have a beamwidth of 90° and a minimum gain of 6db within its beam. To achieve coverage in every direction will require six antennas mounted on mutually perpendicular axes on the spacecraft. As shown in figure 78 four antennas will be mounted on the side of the spacecraft while two others are required for the top and bottom of the spacecraft. The antennas will be flush mounted on the spacecraft.

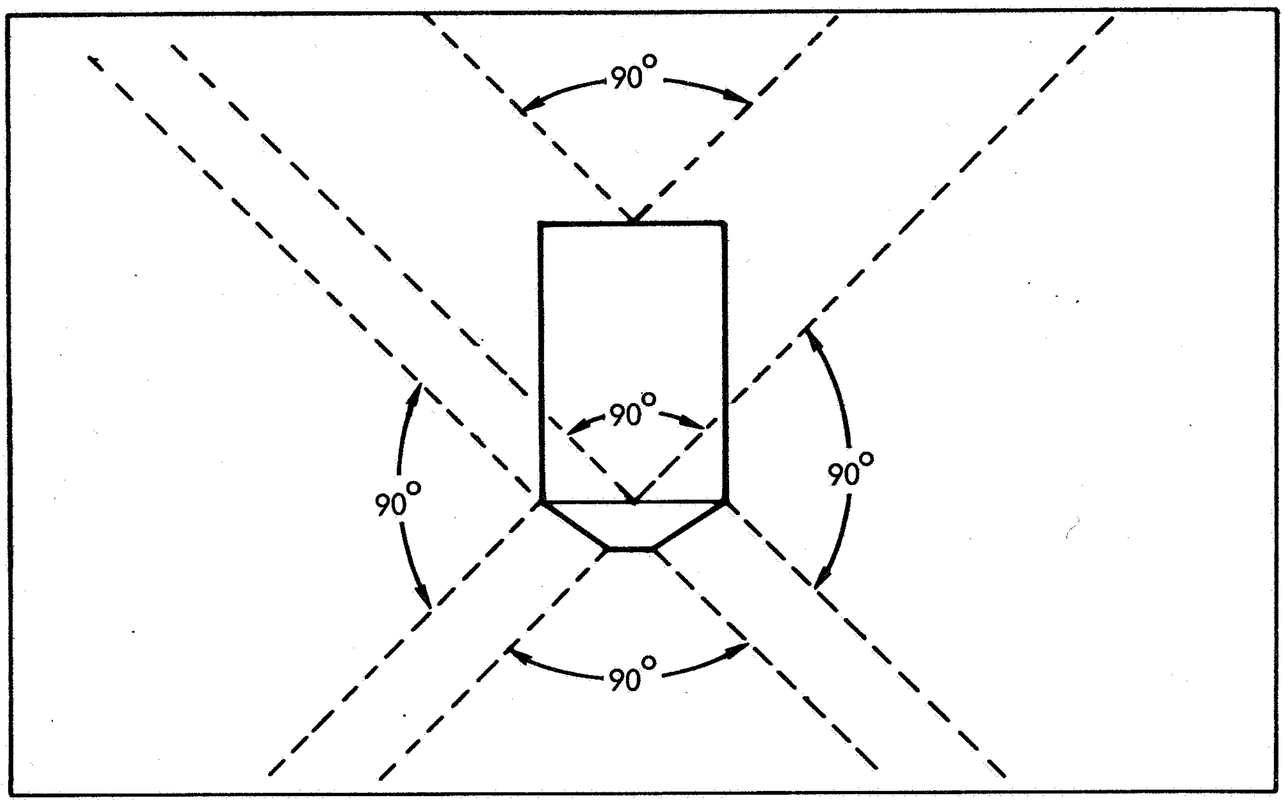


Figure 78. - Omnidirectional television antenna coverage

The operation of the antenna decision circuitry, figure 79, will now be explained. The signals received on each of the six antennas will be detected, sequentially sampled, converted to digital form, and read into shift registers. The output of the register for the first channel will be read into another register called the data register; succeeding register inputs will be read into the data register only if the output of that register is greater than the value stored in the data register. The counters and associated logic circuitry keep track of which channel had the largest signal. At the end of the interval during which all six registers are sampled, the appropriate antenna is used for

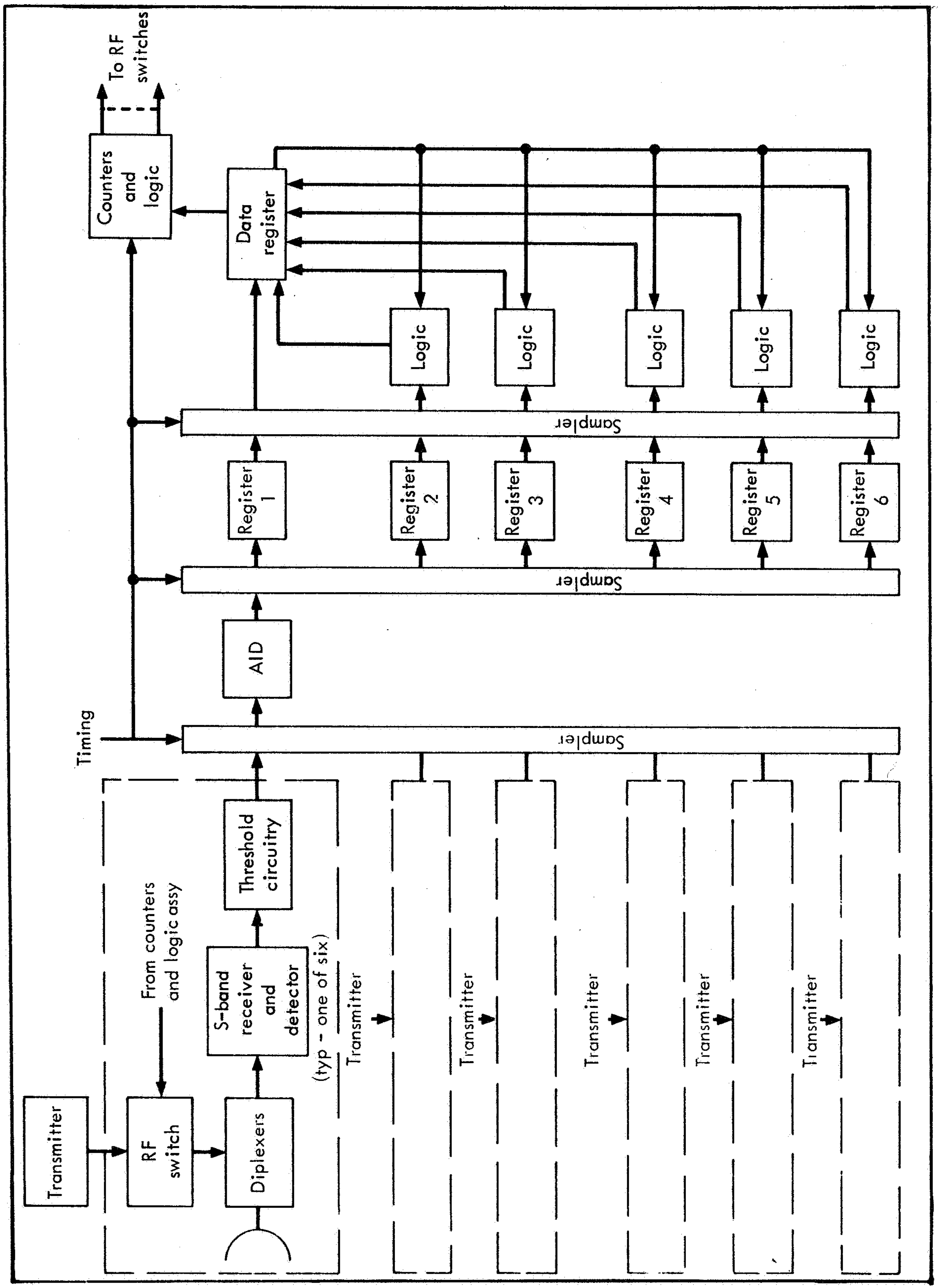


Figure 79. - Tevevision antenna decision - block diagram

transmitting and the remaining five antennas are shut off. The sampling interval is repetitive. The threshold circuitry is included so that reception is possible through any antenna in the absence of an input signal.

The simplest type of antenna for the television channel is a simple open-ended circular waveguide. With the proper choice of waveguide diameter to wavelength ratio, the beam can be made to have a 90° beamwidth. Figure 80 shows such a radiation pattern. Such an antenna may be flush, or nearly flush, mounted on the spacecraft. The minimum length of waveguide required is that length which is required to set up the wave in the proper propagation mode in the waveguide. A waveguide 15 inches long would be adequate, however, structural considerations preclude the use of this long of an antenna.

Another type of antenna which could be used is the cavity backed archimedes spiral. The advantage of this type of antenna is that the front to rear dimension is shorter than the open-ended circular waveguide. Unfortunately, efficiency is only 50%. This antenna is normally used where broad bandwidth and moderate VSWR is required.

Since broad bandwidth is not required for the Primate Spacecraft communications link, the best antenna would be a T-bar fed slot antenna, or similar type. A T-bar slot antenna for this application could be enclosed in a volume 2' x 3" x 2" and mounted flush with the spacecraft exterior.

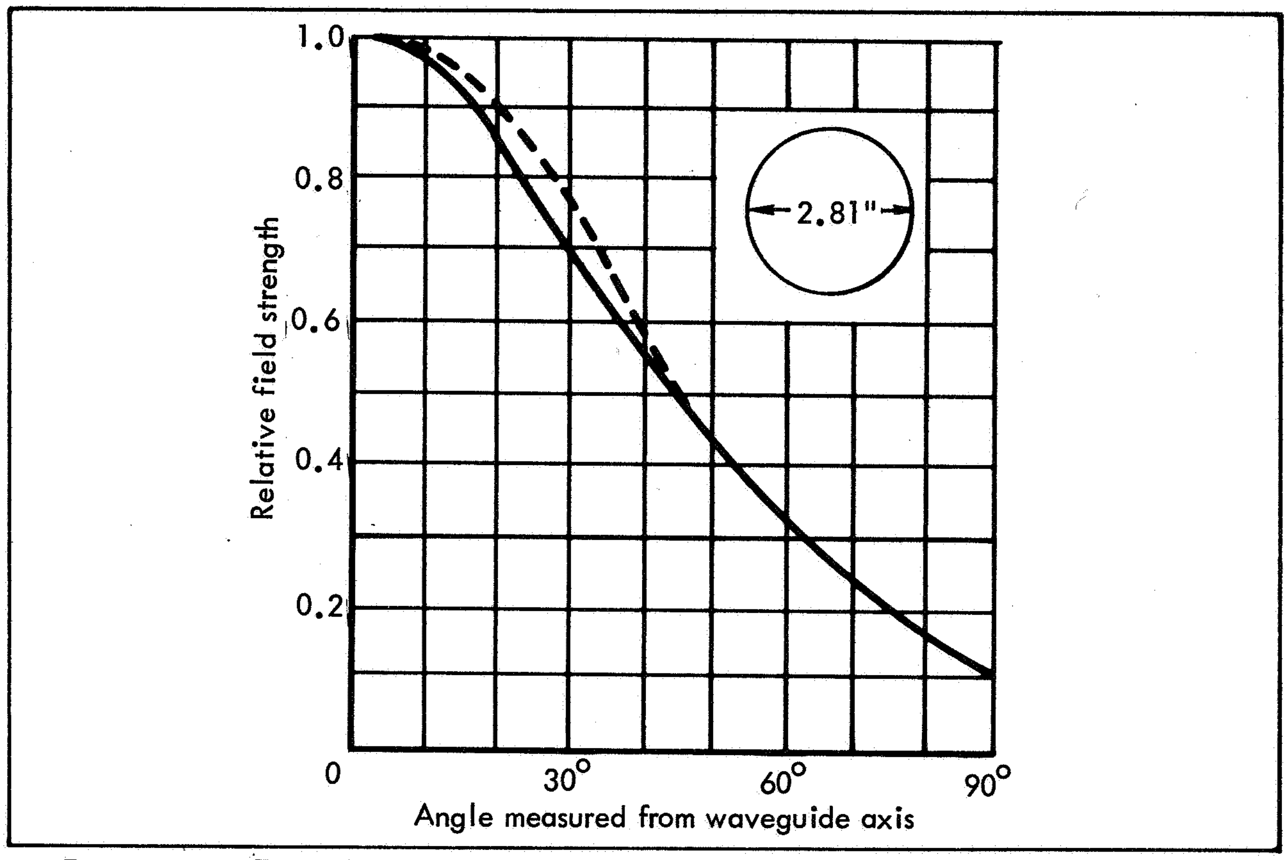


Figure 80. - Typical radiation pattern open ended circular wave guide (free space)

Advance development areas: Following a survey of current suppliers of space qualified magnetic tape recorders, a true video tape recorder employing rotary magnetic heads, was discovered that has been partially space qualified for Marshall Spaceflight Center, Huntsville, Alabama. This recorder was developed by RCA, Defense Electronics Products, or NASA, MSFC under Contract NAS 9-4629.

It is therefore proposed to utilize "Model SH-L-1" of the RCA Video Space recorder for the Primate mission, providing that the recorder is capable of completing the space qualification testing now under way at MSC, Houston.

An outline of the specifications of the RCA video recorder, Model SH-L-1, is listed in table 48.

TABLE 48. - RCA VIDEO RECORDER SPECIFICATIONS

Dimensions	10 x 14 x 6.1 inches
Volume	850 cubic inches
Weight	30 pounds
Bandwidth	dc to 500 KHz
Tape width	1.0 inch
Tape speed	1.25 in./sec.
Rewind time	8 minutes
Head-to-tape speed	140 in./sec.
Singal-to-noise ratio	38 db PP to RMS
Start time	30 seconds
Auxiliary tracks	one audio or digital (2500 bps)
Time stability	at least one part in 10^3
Input/output signal	1.5 volt PP @ 50 ohm
Line voltage	28 ± 4 volts
Power dissipation	50 watts
Available recording time	4 hours
Recording principle	Helical scan, 2 heads, @ 180°
Differential gain	Within 10%

To provide sufficient reliability and storage capacity, it is planned to use two video space recorders with either automatic changeover, or command changeover.

Some additional development will be required to provide voice recording of 50 to 12,000 Hz.

Preliminary equipment list: Table 49 lists the major equipment elements for the telemetry subsystem.

Command and Control

The Primate Spacecraft Command and Control Subsystem consists of a command decoder to decode ground commands and a programmer/sequencer to provide spacecraft subsystems with timing and control that can be modified by up-link commands.

The approach presented herein evolved from the selected approaches of the centralized vs. Remote and Data Encoding Trade Studies (ref 6).

A simplified block diagram of the Command and Control Subsystem is shown in figure 81.

Command and control requirements and constraints. - The Command and Control Subsystem decodes the commands received and demodulated by the telemetry subsystem. Using these commands, the Command and Control Subsystem regulates all on-board timing and control signals for the Electrical Power, Telemetry, Instrumentation, Life Support, Thermal Control, Attitude Control, and Structure and Mechanical Subsystems.

The Apollo PCM two-tone format will be used for spacecraft commands and the format will be the same as that used for the Apollo Command Module, except for necessary vehicle and subsystem address changes.

The Primate Spacecraft will also have the ability to receive commands through the GSE launch tower umbilicals when it is on the launch pad. All monitoring prior to launch will be accomplished using the launch tower GSE umbilical.

The Command and Control Subsystem will operate throughout the functional life of the Primate Spacecraft through prelaunch, launch-to-separation, post-separation (orbital), and recovery modes.

The prelaunch phase of spacecraft operation is initiated two days before launch and terminates at booster lift-off. At T minus two hours, spacecraft switches to internal power. During the prelaunch mode the Command and Control Subsystem becomes self-sufficient with provision for a command override through the GSE umbilical.

The launch-to-separation phase of spacecraft operation starts with booster lift-off and ends when the Command Service Module releases the spacecraft in orbit. During this phase, Command and Control Subsystem operation is entirely self-sufficient.

TABLE 49. - PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested manufacturer	Part No.	Quantity per spacecraft
1	TV and voice recorder	RCA SH-L-1		2
2	Switches (circuitry)	Northrop		5
3	PCM encoder	Radiation Inc.		1
4	Data recorder			2
5	Premodulation processor	Collins	Item 5 corresponds to simplified version of Collins 960C-1 premodulation processor	1
6	(6.5 kb) buffer storage	Telemetermagnetics		1
7	Dual transponder	Motorola	Similar to Collins 621F	1
8	TV transmitter	Watkins, Johnson, Hughes		2
9	Telemetry transmitter	Watkins, Johnson, Hughes		2
10	Diplexer, circulator TV assembly	Rantec		1
11	Diplexer, circulator data assembly	Rantec		1
12	Data antennas	Northrop		2
13	TV antennas	Northrop		6

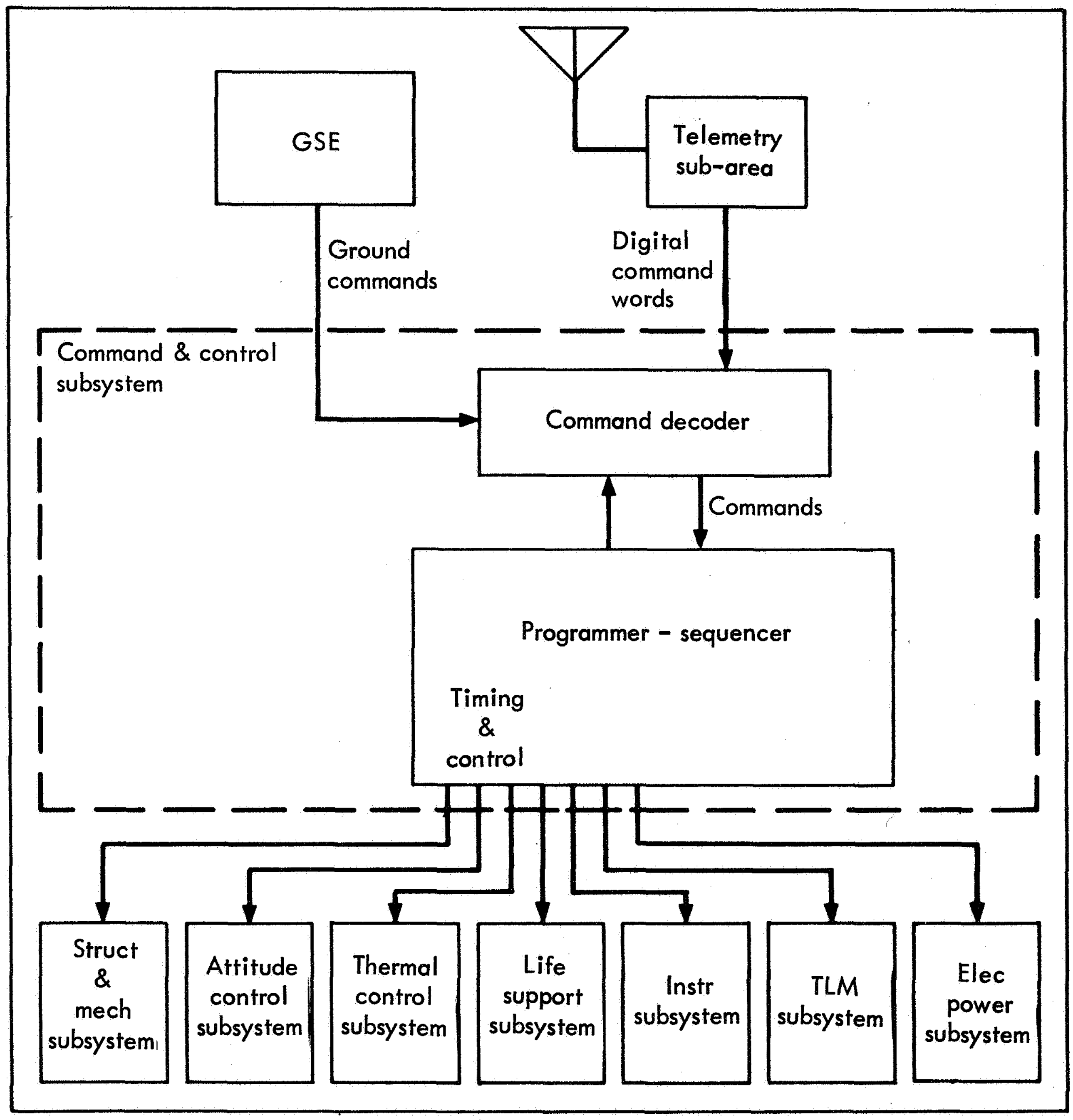


Figure 81. - Command and control subsystem - simplified block diagram

The orbital phase of spacecraft operation, lasting approximately one year, starts when the Command Service Module injects the spacecraft into its final orbit and ends at the start of the recovery mode.

The recovery phase of spacecraft operation begins when the spacecraft is docked with the Command Service Module and ends when the KILL command becomes effective. The "KILL" command disconnects the solar panels from the spacecraft power subsystem thereby extinguishing the communication ability of the spacecraft in approximately three hours. Terminating the communications link will permit subsequent use of similar (or exact) commands by other spacecraft without the interference from a responsive Primate Spacecraft. Again, operation of the Command and Control Subsystem is self-sufficient throughout the recovery phase with provision for modification by ground command. Table 50 summarizes the commands from the ground station to the spacecraft via up-link telemetry.

Command and control system functional description and performance. - As stated earlier, the command and control subsystem provides the timing and control signals for all of the various spacecraft subsystems. Figures 82 and 83 illustrate the prelaunch and launch-to-separation phases command and control operation time lines, while figures 84 and 85 show the sampling intervals of various data points during orbit. Table 51 itemizes the number of data points and contributing subsystems, while figure 86 represents a typical profile of television and data dump pass.

To sample and transmit the accumulated data, the following assumptions are made:

(1) The spacecraft will complete 10 consecutive orbits without ground coverage.

(2) Six consecutive orbits will be required to dump the data.

(3) The total dump time per day will be 38 minutes.

(4) The dump time will range from four minutes to 9 minutes over the ground station per pass.

(5) One minute acquisition time will be required for each dump orbit.

(6) The random and more frequently occurring data will be temporarily stored and sequenced onto the tape at an opportune time. For example, EKG is sampled for five minutes for each Primate at four hour intervals.

The time required to dump the data accumulated during 10 orbits, is approximately 8 minutes for a 51.2 Kbps rate, however, a total of 38 minutes is available over site for 6 orbit passes. Assuming that 1 minute acquisition time is required for each pass, 32 minutes are available to dump 8 minutes of accumulated data, leaving 24 minutes for real-time transmission. The real-time transmission will be on command following the down link dump. However, override by ground command for other programmed sequence is also possible.

Transmission of television data is independent of other data transmission; however, it is simultaneously dumped with other spacecraft data. Real-time or television recorder dump is a commanded function and will usually be initiated following an acceptable status of the quick-look engineering data. However, it may be advisable, due to the large quantity of television data, to start television transmitters as soon as possible upon acquisition and then stop the transmitters if the quick-look engineering data is not favorable. The television profile is shown in figure 86.

TABLE 50.- COMMAND REQUIREMENTS

Subarea	No.	Name	Critical	Mission Phase				Remarks
				PL	L	O	R	
LIFE SUPPORT ECU	C1	Main Fan Index		X		X		Switch to Next Main Fan
	C2	Contam. Cont Fan/Index		X		X		Switch to Next Contaminant Cont. Fan
	C4	Freon Pump Index		X		X		Switch to Next Freon Pump
	C5	LC Evacuate	X				X	Depressurize Life Cell - Astronaut Source
	C6	Recharge Freon				X		Recharge Freon Loop
	C7	LC Purge	X			X		High Leak Rate
Life Cell	C8	Wall A Extend		X		X		
	C9	Wall A Retract		X		X		Move Wall (May be astronaut source)
	C10	Wall B Extend		X		X		
	C11	Wall B Retract		X		X		
	C12	Release Wall A				X		Disconnect Drive From Wall
	C13	Photoperiod Continuous				X		
	C14	Photoperiod Normal				X		Change Photo Period
	C15	Release Wall B				X		Disconnect Drive from Wall
Feeder	C18	Wall Charge A		X		X	X	Turn Wall Charge ON or OFF (toggled)
	C19	Wall Charge B		X		X	X	(may be astronaut source)
	C24	Food Override A				X		Eliminate Behavioral Task
	C25	Food Override B				X		Requirement for Reward
Waterer	C28	Water Override A				X		Eliminate Behavioral Task Requirement for Reward
	C31	Water Override B				X		Eliminate Behavioral Task Requirement for Reward
Behavioral Panel	C32	Activate B.P.A.		X		X	X	Control Status of B.P.
	C33	Deactivate B.P.A.	X	X		X	X	(may be astronaut source)
	C34	Time Interval Inc. A		X		X		Required by P.I. Change
	C35	Time Interval Dec. A		X		X		Work Task
	C40	ILK Response Inc. A		X		X		
	C41	ILK Response Dec. A		X		X		
	C42	Activate B.P. B		X		X	X	Control Status of B.P.
	C43	Deactivate B.P. B	X	X		X	X	(may be astronaut source)
	C44	Time Interval Inc. B		X		X		
	C45	Time Interval Dec. B		X		X		Required by P.I. Change
	C50	ILK Response Inc. B		X		X		

TABLE 50. - (Continued)

Subarea	No.	Name	Critical	Mission Phase PL	L	O	R	Remarks
Behavioral Panel (cont'd)	C51	ILK Response Dec. B		X		X		Required by P.I. Change Work Task
	C96	Exc. Force A Inc.		X		X		
	C97	Exc. Force A Dec.		X		X		
	C98	Exc. Force B Inc.		X		X		Control exerciser force and extension into Life Cell
	C99	Exc. Force B Dec.		X		X		
	C100	Retract Exc.				X	X	
	C101	Vig. Time A Inc.		X		X		
	C102	Vig. Time A Dec.		X		X		
	C103	Vig. Time B Inc.		X		X		Change behavioral task
	C104	Vig. Time B Dec.		X		X		
Recovery Capsule	C52	Load Primate A	X				X	Start Sequence
	C53	Evacuate RC A	X			X		Dry Cadavar
	C54	Load Primate B	X				X	Start Sequence
	C55	Evacuate RC B	X			X		Dry Cadavar
	C56	INCR Door Open A					X	
	C57	INCR Door Open B					X	Provide Close Control of Closing Door
	C58	INCR Door Close A					X	(may be astronaut source)
	C59	INCR Door Close B					X	
Mass/Volume Measurement Device	C60	M/VM Task on A		X		X		Makes M/VM Task part of behavioral task
	C61	M/VM Task on B		X		X		
	C62	M/VM Task off A		X		X		Eliminates M/VM Task from schedule
	C63	M/VM Task off B		X		X		also, opens door
THERMAL CONTROL	C64	Radiator Index				X		Switch to alternate radiator
STRUCTURE and MECHANICAL	C65	Insertion Seq. Backup	X		X			Backup automatic functions (may be astronaut source)
	C105	Deploy Antennas			X			Deploy all antennas
	C106	Deploy Solar Panels			X			Deploy all solar panels
INSTRUMENTATION	C66	TV A On		X		X	X	Turn on TV upon Request
	C67	TV A Off		X		X	X	
	C68	TV B On		X		X	X	When over ground station
	C69	TV B Off		X		X	X	(may be astronaut source)

TABLE 50. - (Continued)

Subarea	No.	Name	Critical	Mission Phase PL	L	O	R	Remarks
INSTRUMENTATION (con'td.)	C70	Index TV Window A		X		X	X	Provide clean viewport in case of fouling
	C71	Index TV Window B		X		X	X	(may be astronaut source)
	C72	Startle Noise		X		X	X	Test primate response
	C107	TVA Turret Change		X		X	X	Change from wide angle to narrow angle lens
	C108	TVB Turret Change		X		X	X	
	C109	TV Secondary Recorder		X		X		Command secondary TV recorder
	C110	Calibration		X		X		Calibrate PCM and FM/FM subsystems
TELEMETRY	C73	TV Recorder Play-back		X		X		Retrieve information on request
	C74	Data Recorder Play-back		X		X		
	C75	Spare		X		X	X	Location of S/C
	C76	Spare		X		X	X	(may be astronaut source)
	C77	MSFN On				X	X	Turn transmitters on
	C79	XMIT Off		X		X	X	Turn transmitters off
	C112	TV Transmitter 2 On		X		X		Switch to alternate transmitter
	C113	Step TV Antenna		X		X	X	Switch antennas
COMMAND and CONTROL	C80	Real Time Data		X		X	X	Real Time Transmit Data
	C81	Real Time TV		X		X	X	Real Time Transmit TV
ELECTRICAL POWER and CABLING	C82	Turn Tel On		X		X	X	Control power consumption
	C83	Turn Tel Off		X		X	X	as required
	C84	Turn AC On	X	X		X	X	Interlock with Docking
	C85	Turn AC Off	X	X		X	X	(may be astronaut source)
	C86	Solar Panel Disconnect	X			X	X	Kill S/C after mission
	C87	Solar Panel Connect	X			X	X	
	C114	Battery Select SW		X		X		Switch from primary battery
	C115	Charge Reg. 1 SW		X		X		
	C116	Charge Reg. 2 SW		X		X		Disconnect charge regulator
	C117	Charge Reg. 3 SW		X		X		
	C118	Charge Reg. 4 SW		X		X		
	C119	Invert Tran. Ovrde 1		X		X		
	C120	Invert Tran. Ovrde 2		X		X		Disconnect inverter-override automatic switching
	C121	Invert Tran. Ovrde 3		X		X		
	C122	Reg. Tran. Ovrde		X		X		Override automatic switching
	C123	Non-ess. Load Disconnect		X		X		Disconnect non-essential load

TABLE 50. - (Concluded)

Subarea	No.	Name	Critical	Mission Phrase				Remarks
				PL	L	O	R	
ATTITUDE CONTROL	C88	Torque Gyro X Plus		X		X	X	Emergency Torquing of Gyro to control attitude
	C89	Torque Gyro X Minus		X		X	X	
	C90	Torque Gyro Y Plus		X		X	X	
	C91	Torque Gyro Y Minus		X		X	X	
	C124	Torque Gyro Z Plus		X		X	X	
	C125	Torque Gyro Z Minus		X		X	X	
	C126	Pos. Roll Jet On				X		Direct Jet Control to control S/C attitude
	C127	Neg. Roll Jet On				X		
	C128	Pos. Pitch Jet On				X		
	C129	Neg. Pitch Jet On				X		
	C130	Pos. Yaw Jet On				X		
	C131	Neg. Yaw Jet On				X		
	C132	Roll Gyro Cage	X	X		X		Provide a torquing capability for drift compensation
	C133	Roll Gyro Uncage		X		X		
	C134	Pitch Gyro Cage	X	X		X		
	C135	Pitch Gyro Uncage		X		X		
	C136	Yaw Gryo Cage	X	X		X		
	C137	Yaw Gyro Uncage		X		X		

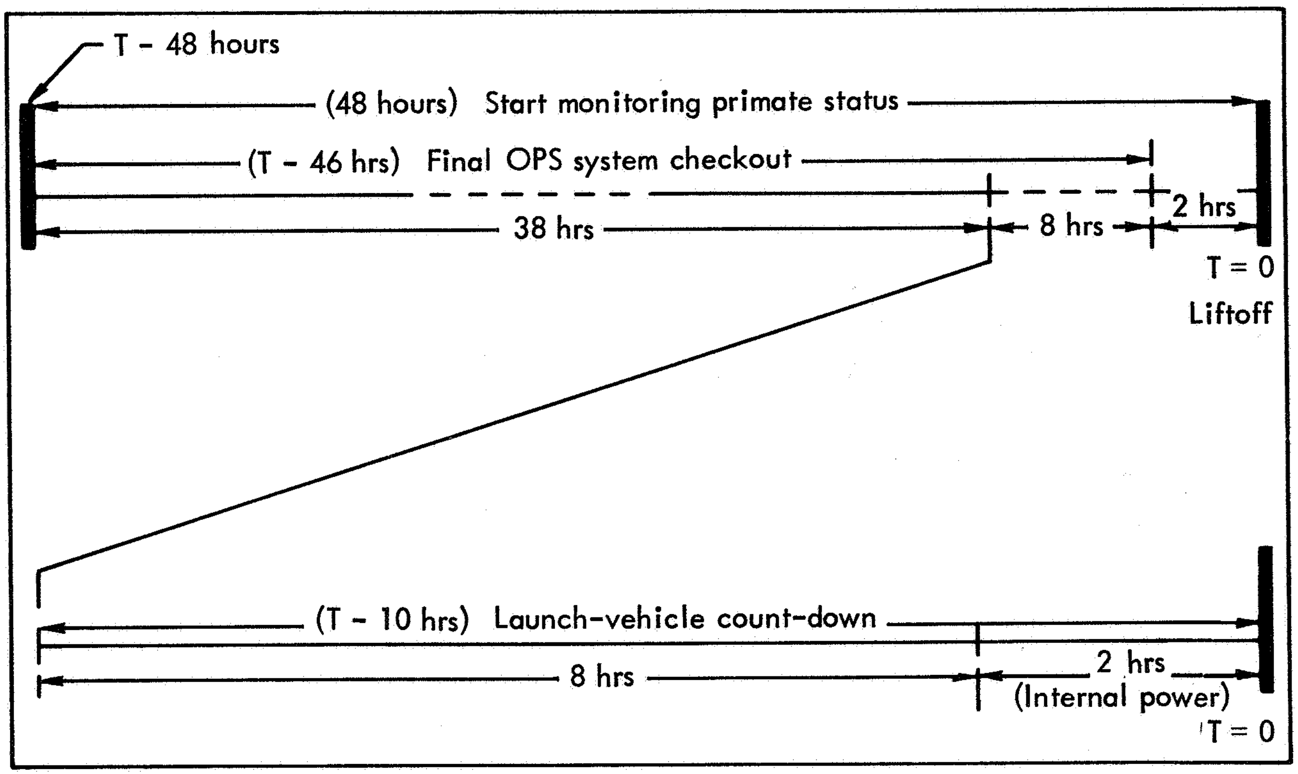


Figure 82. - Pre-launch profile

The interval of television recording for both of the primates is as follows:

(1) One minute at the beginning of each hour

(2) One minute before lights-on

(3) One minute after lights-off

(4) One minute at the middle of the 10 hour dark period

The real-time television transmission occurs for one minute, 30 seconds per primate, at each pass for six orbits. It will not be necessary to record the real-time television on the spacecraft. To more efficiently utilize the dump time, the real-time television will stop the television recorder dump at the beginning of its one minute period and start the television recorder dump upon completion of the one minute real-time transmission. This timing as well as all other timing will originate from the Programer/Sequencer.

Command decoder: Input from the telemetry demodulator to the command decoder is coded video. The decoder accepts or rejects commands after examining the word format and spacecraft indentification characteristics of the uplink command. Figure 87 is a block diagram of the decoder, and figure 88 illustrates the uplink command structure.

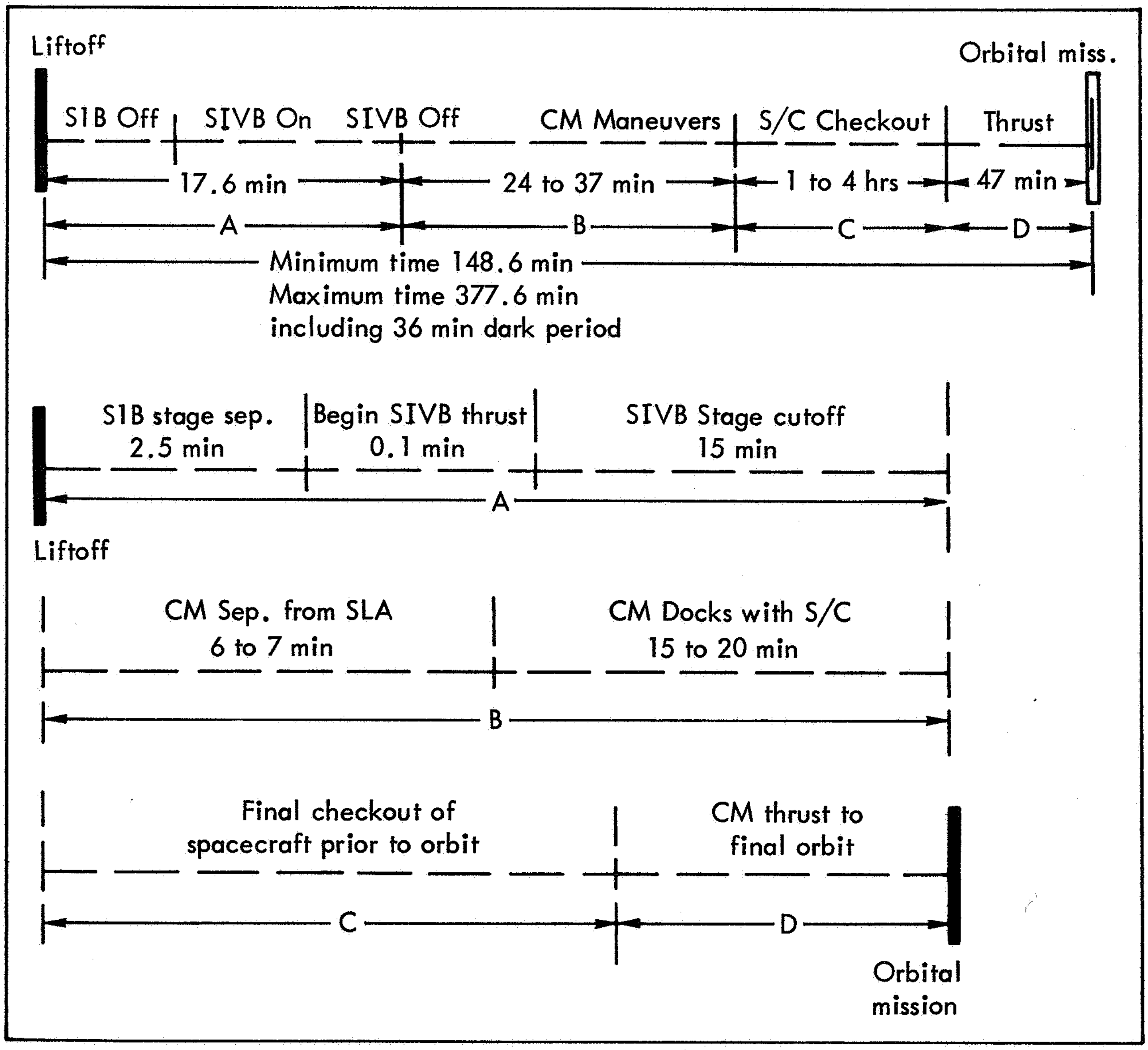


Figure 83. - Launch-to-separation profile

Commands sent to the spacecraft consist of 12 bits; the first 3 bits determine the vehicle address. A command is sent four times in series to the spacecraft. The decoder accepts or rejects the command depending on the command length and vehicle address as determined by the word validity register. The valid command is then transferred, through a data switch, to one of four command comparator registers. The second of the four identical up-link commands is received in the word validity register and the process repeats until all four commands have been transferred to the command comparator registers. If all four commands are identical, a verification response is transmitted to the ground site.

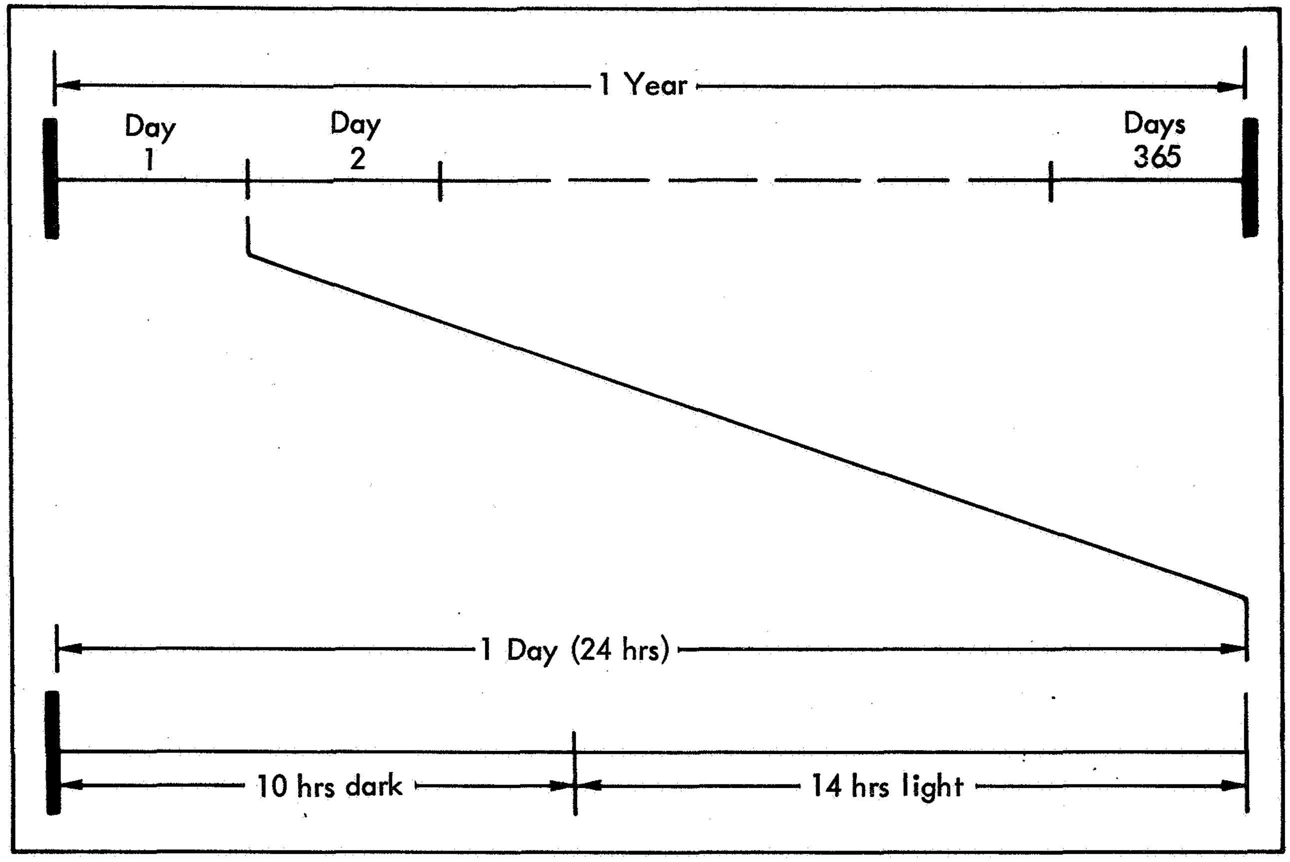


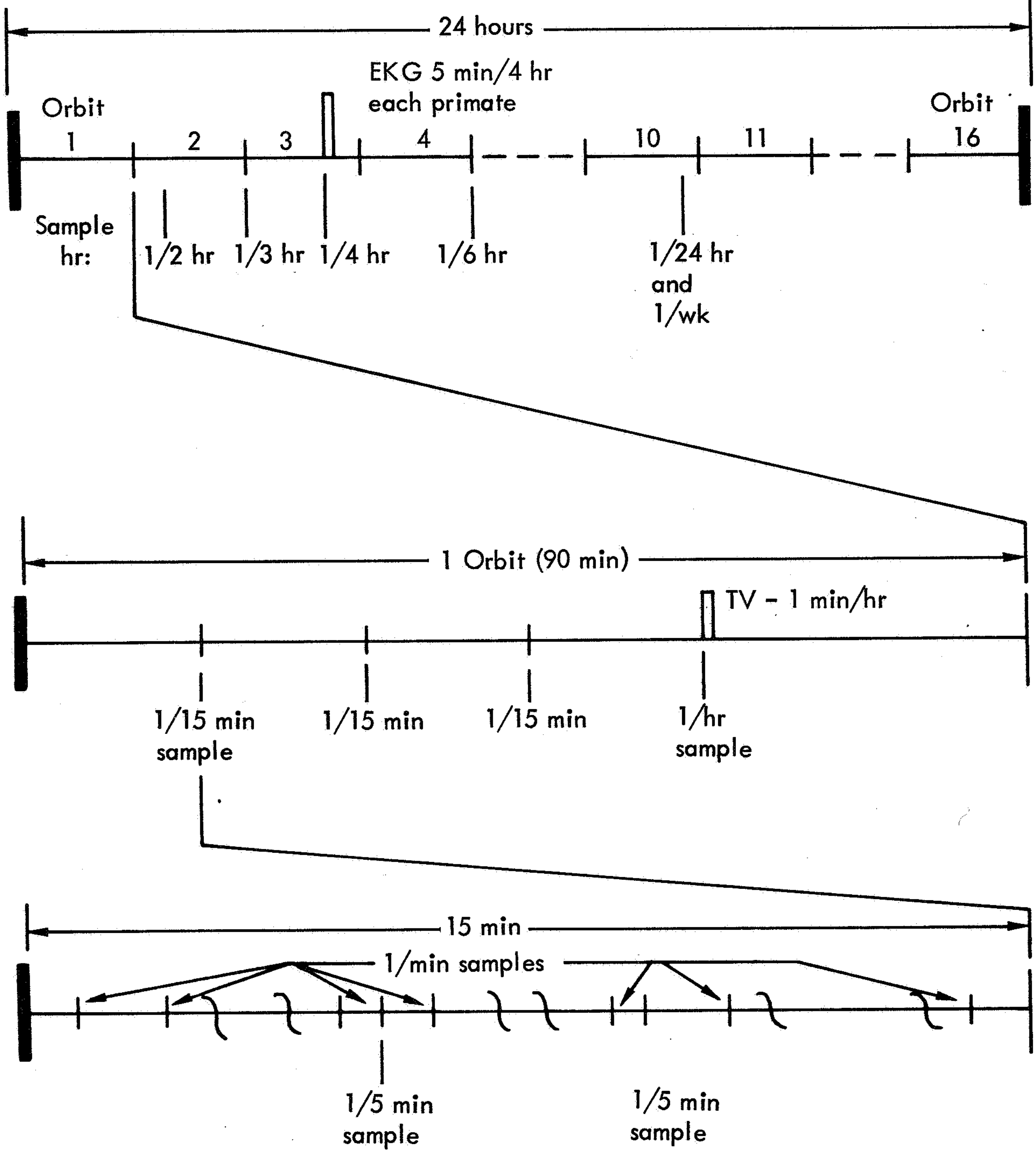
Figure 84. - Typical mission day profile

Once the vehicle address has been established, the three bits are no longer significant and are not transferred to the command comparator register. After the command has been verified as correct, it is sent to the programmer sequencer for execution. Of the four identical commands only one is sent to the programmer sequencer; the other three are erased. The comparator register is then free to accept additional up-link commands.

Commands sent to the spacecraft have the same format and length as used by the Command Service Module for the Apollo mission. The Apollo command structure is shown in figure 88. Due to the large number of commands required for the primate spacecraft, it is proposed to employ the three system address bits as part of the six command bits. This increases the up-link command capability to nine bits from six bits.

Programmer sequencer: After the decoder verification has been satisfied, the commands are transferred to the programmer sequencer storage from which the command execution commutator executes the command. A block diagram of the programmer sequencer is shown in figure 89.

The programmer sequencer has provisions to store up to four uplink commands. To avoid timing interference that could be caused by an immediate



See Table 51 for the number of data points at these sample periods

Figure 85. - Typical data collecting orbit profile

TABLE 51. - SPACECRAFT DATA POINTS SUMMARY

	Sample Interval (hours)											
Subarea	1/wk.	1/24	1/6	1/4	1/3	1/2	1/1	4/1	12/1	60/1	continous	as occurs
LIFE SUPPORT	12	19	12		2		3	1	7		12	12
BEHAVIORAL PANEL											34	
THERMAL CONTROL	3	2	2									
MECH. & STRUCTURAL												9
INSTRUMENTATION		1	2	2			2		4	2		
TELEMETRY			13									
COMMAND & CONTROL		8										
ATTITUDE & CONTROL		3				13						
ELEC. POWER	1	7	2				19			2		

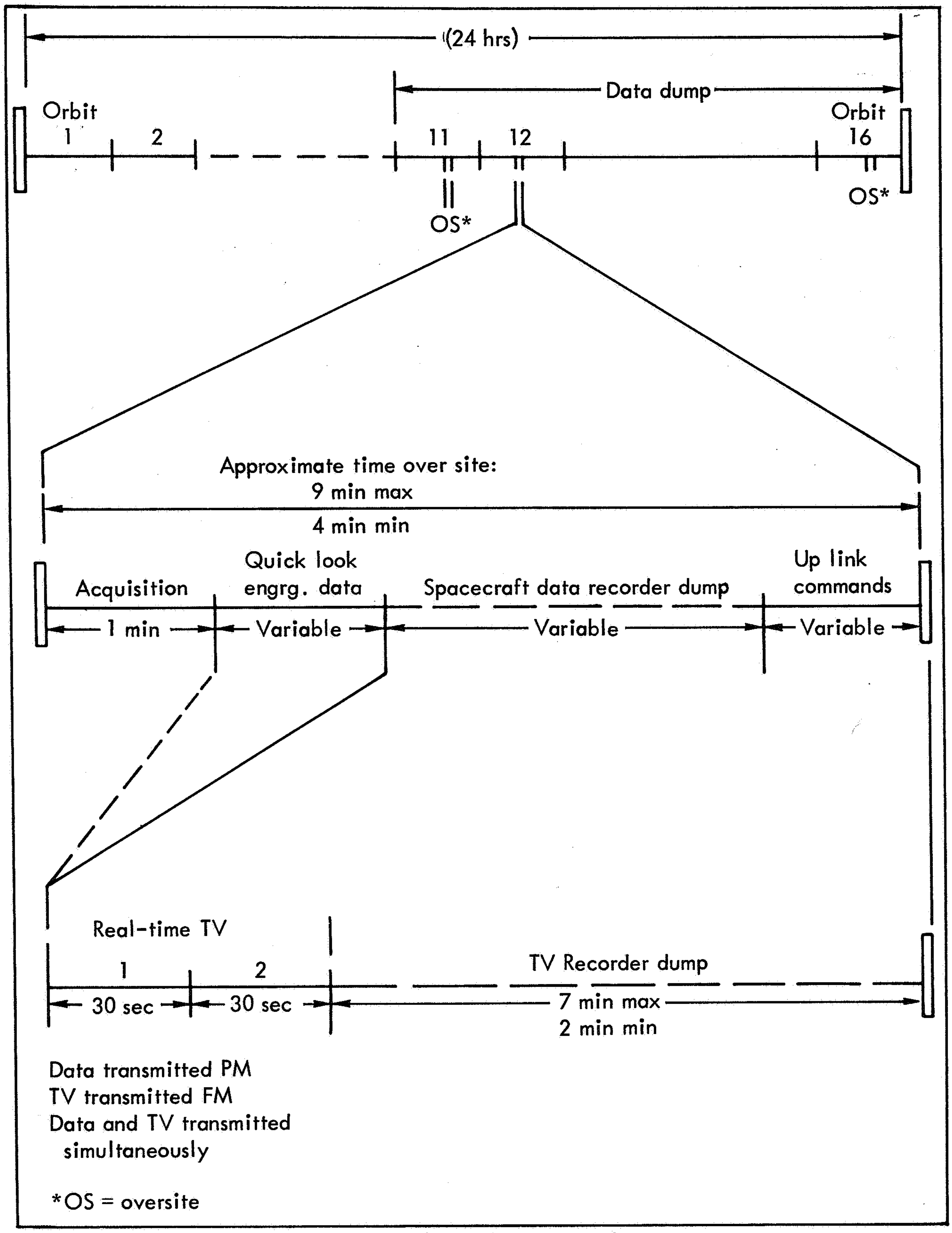


Figure 86. - Typical dump profile

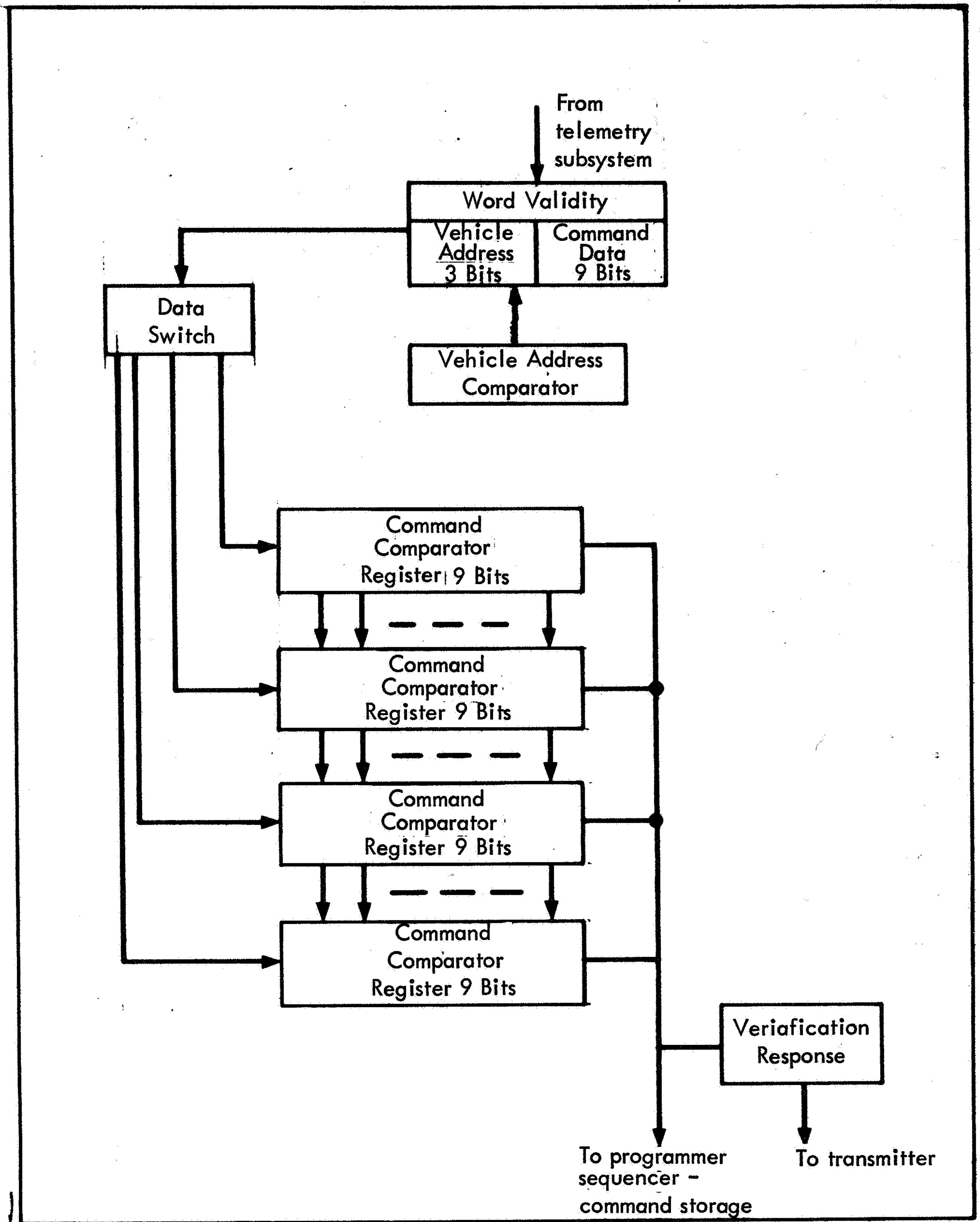


Figure 87. - Decoder block diagram

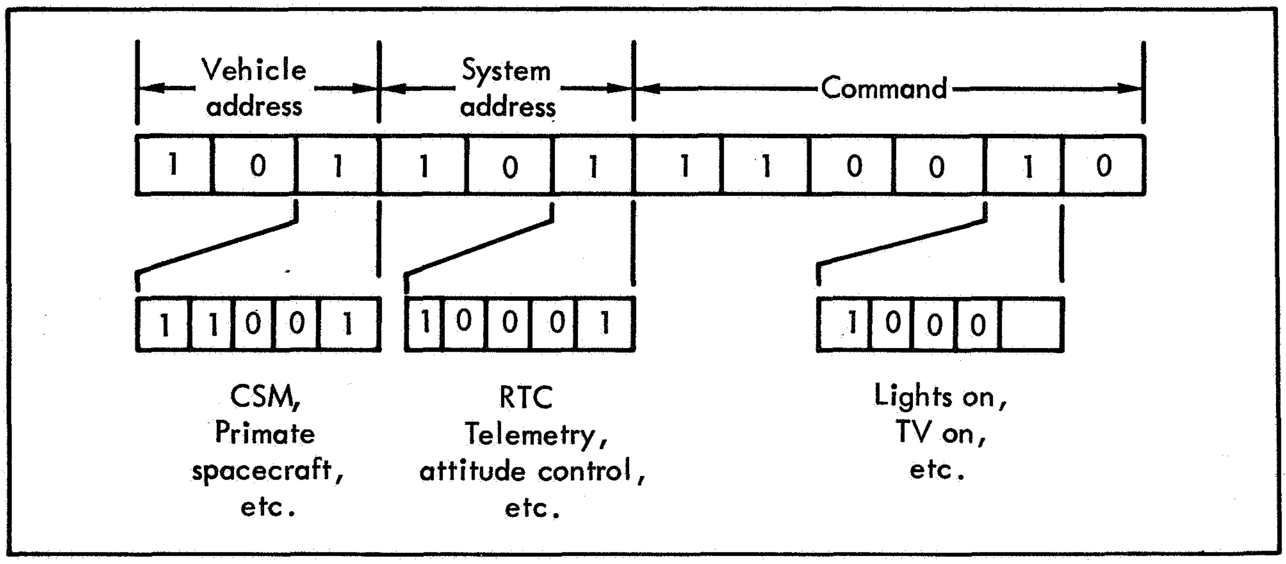


Figure 88. - Up link command structure

execution of a command, the command remains in the storage until read out by the command execution commutator. The four command storage cells are numbered one to four, the read-in ot commands are stored in the smaller numbers first and progress to the higher. Also, read-out of the commands progresses from the smaller numbered cells and to the higher. Interlace exists in the storage such that simultaneous read-in and read-out can be implemented.

Generally, the execution time is less than the receiving, verification and storage of a command time. However, considerable time may elapse in obtaining specified results. For example, recorder playback would require several minutes.

Basically there are two types of stored commands. Commands for immediate effectiveness and commands to become effective at a later time. Commands for immediate execution would include but not be limited to the following:

(1) Override task completion for food and water
(2) Turn ON or OFF the behavioral panel
(3) Startle response
(4) Real-time television or other data
(5) Start playback of recorders
(6) Critical engineering data.

Commands to become effective at a later time would include but not be limited to the following:

(1) Vary the sequence of task

(2) Modify the characteristics of the task

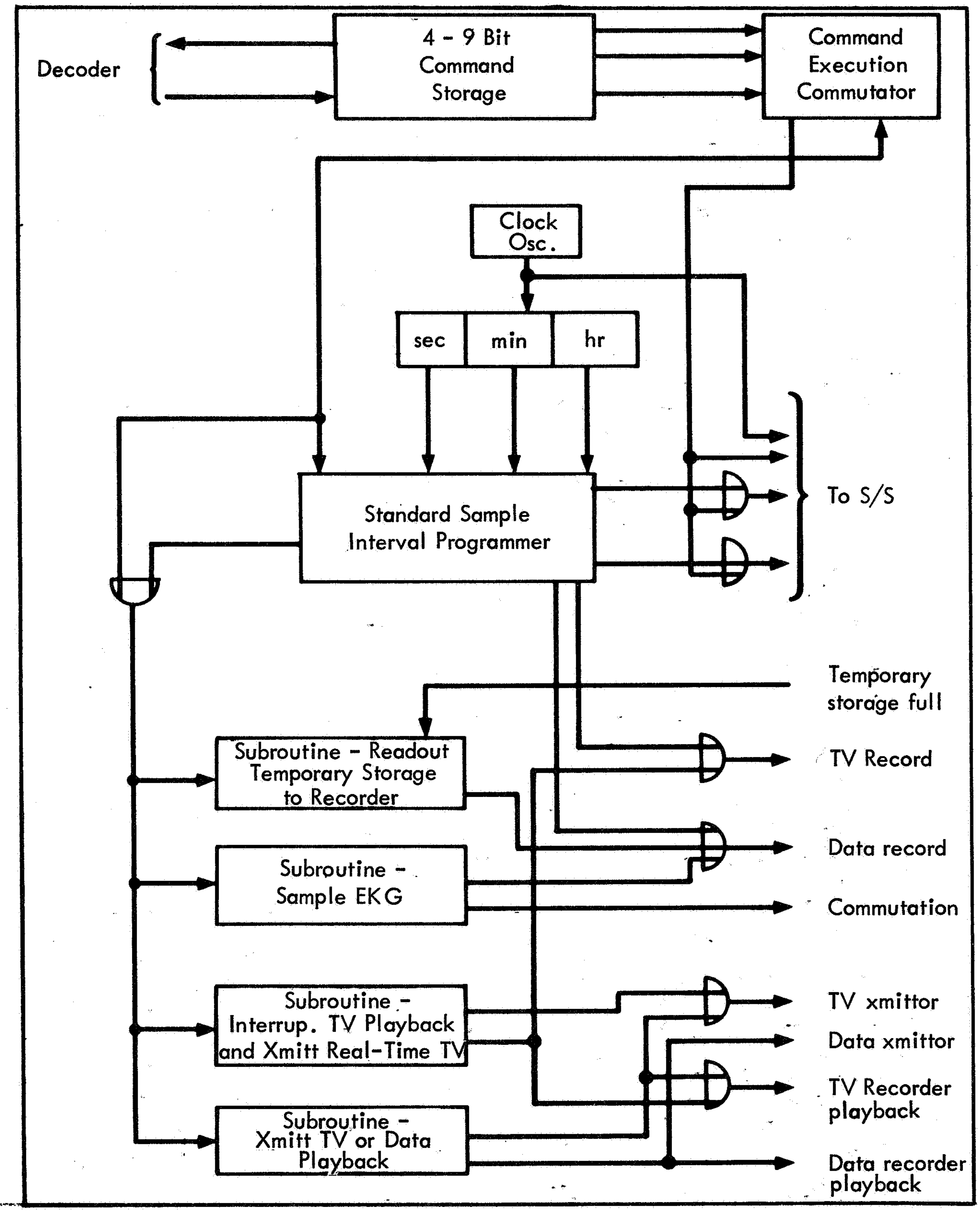


Figure 89. - Program sequencer block diagram

(3) Change the frequency of recording intervals of the recorders (television (or data recorders).

In addition to the four commands, four subroutines are also stored in the programmer sequencer to improve efficiency in sequencing and storing on-board data. Routines for control and gathering of data also include but not be limited to the following:

(1) Initiate and sequence data blocks to the recorder at periodic recording intervals

(2) Sequence data to the recorder at other than the standard record interval if overflow of the temporary storage is evident

(3) Playback of television or data recorders.

The programmer-sequencer performs synchronization of all onboard data collecting and spacecraft control, including such functions as:

(1) Start and stop recorders and transmitters
(2) Interrogate various sub-areas at predetermined intervals
(3) Provide override capability upon ground command
(4) Monitor engineering status of sub-areas
(5) Energized redundant features when necessary
(6) Shift data from temporary storage to the tape recorder
(7) Gather data into a suitable format for block recording.

On-board subroutines are used when necessary to provide several timing or control signals which are required to complete an operation. The four subroutines in the programmer sequencer are to accomplish the following form tasks.

First, the temporary storage contents are transferred onto the magnetic tape recorder. This could occur at a standard record cycle or be forced due to storage overflow. The subroutine is energized by the standard sample interval programmer or by the overflow status of the storage.

Secondly, the EKG is sampled every four hours for five minutes. This subroutine provides timing and control to start the commutator and recorder for recording the EKG sample. The EKG is sampled for one primate followed by the sample from the other. It is possible to transfer the temporary storage to the record between the EKG samples if it is evident overflow would occur prior to completion of the following EKG sample. Much of the data must be annotated with respect to the time of day. The Behavioral Panel requires ten milliseconds resolving accuracy of time. The programmer-sequencer provides the necessary timing signals to accomplish this order of resolution. The programmer-sequencer must also provide signals to energize various redundant features should malfunctions arise.

Thirdly, when real-time television transmission is commanded during television recorder playback, a subroutine provides the timing and control signals to interrupt the playback of the recorder and energized the correct camera for transmission.

Fourth, a subroutine energizes the recorders and transmitters for commanded playback of television, data or both. Up-link command for playback overrides on-board recording of television or data if television recording has not yet begun its record cycle or if temporary storage is not being transferred to the tape record. If the television recorder is in its record cycle or the data recorder is collecting the contents of the temporary storage they continue until completion, before accepting a playback command.

The onboard time of day clock is a crystal controlled divider, presenting the time from one millisecond to 24 hours. The error of the spacecraft clock can be determined by the ground station at data dumps; therefore, updating of the onboard clock in not necessary.

Uplink commands are required for both data and television recorder playback. Initiation of Real Time transmission of data or television requires up-link commands. Timing to sequence and control recorders and transmitters originates within the programmer-sequencer.

The standard sample interval programmer performs the required timing and control signals for collecting and processing primate and engineering data. The spacecraft operates in a self-contained fashion, without the aid of ground commands, to accomplish mission requirements. Up-link ground commands are necessary to deviate from the mode of operation dictated by this programmer. After a deviation has been effected by a command, this modification remains effective until modified by a subsequent command.

The behavioral panel contains circuitry to monitor and modify the various primate task as indicated by the commands.

Power switched at the using subsystem is not supplied by the Command and Control subsystem. The power is provided by the electric power subsystem through relays or solid state devices switched by signals originating in the Command and Control subsystem.

All onboard engineering and primate data are gathered in a programmed sequence according to the data requirements. The programmer sequencer interrogates the various subsystems at established sample intervals. Data is read out to the telemetry subsystem where it is processed into suitable formats for eventual downlink transmission. Uplink commands provide the flexibility to override and deviate from this programmed sequence of collecting data.

The behavioral panel electronics perform a sequence of scrambled tasks for the primates work pattern. This scrambled sequence of task presentation can also be modified by uplink commands.

Advance development areas. - Development effort of an advanced or special nature beyond the normally anticipated engineering effort is not required.

Preliminary Equipment Test. - A parts list is shown in table 52.

TABLE 52. - COMMAND AND CONTROL
PRELIMINARY EQUIPMENT LIST

Item	Description	Suggested manufacturers	Part No.	Quantity per spacecraft
1	Decoder	Northrop		1
	12 bit shift registers			1
	9 bit shift registers	Philco		4
	12 bit comparator			1
	4 pos data switch			1
2	Programmer Sequencer	Northrop		1
	clock oscillator	Monitor Elec. Corp.		1
	clock divider			1
	9 bit register	Philco		4
	4 bit register			8
	8 bit register			4

The Primate Spacecraft power subsystem is one of the critical elements upon which the success of the extended weightlessness experiment is dependent. Therefore, the power subsystem must be designed to provide an uninterrupted source of power for the duration of the mission.

The baseline design described herein evolved from the approaches selected from trade studies of the Power Subsystem, System Interaction, and Source and Distribution (ref. 5).

The power system configured for the Primate Spacecraft consists of the following major functional elements:

(1) Power sources including solar array, secondary batteries and primary batteries. Ground power is required during ground operations.

(2) Conversion equipment including regulators and inverters required to produce regulated dc and ac from the raw power bus.

(3) Distribution including the busses, cabling and switching required to provide power to the loads.

(4) Control and protection including the circuitry which provides for automatic and manual control of the conditioning, switching, and storage devices of the power subsystem.

(5) Pyrotechnic circuitry including the Pyrotechnic Control Unit (PCU), Pyrobattery and Pyrotechnic Actuators. This circuitry provides the mechanization for arming and firing the various pyrotechnic devices used for deployment of spacecraft devices.

<u>Electrical power and cabling subsystem requirements</u>. - The Primate Spacecraft power subsystem is mission critical and must be designed for maximum reliability. It is essential that the power subsystem design provide redundancy where failure at the black box or component level would result in serious compromise of mission objectives.

The Spacecraft will be on primary battery power from T-2 hours before launch until solar panel deployment. A power profile representative of the load requirements is shown as figure 90. The assumption has been made that power demand will be minimized during this phase of the mission since the solar array is inoperative. The power requirements for a worst case orbit a are shown as figure 91. A detailed power load breakdown is given in table 53. The analysis assumes 80% inverter and 90% voltage regulator efficiency. A 250 n. mi. circular orbit having a 35.8 minute maximum dark time and a 57.9 minute minimum sunlight time was utilized to size major power subsystem functional elements.

To minimize the chance of catastrophic failure occurring as a result of a malfunction in the power subsystem, redundant major units such as regulators

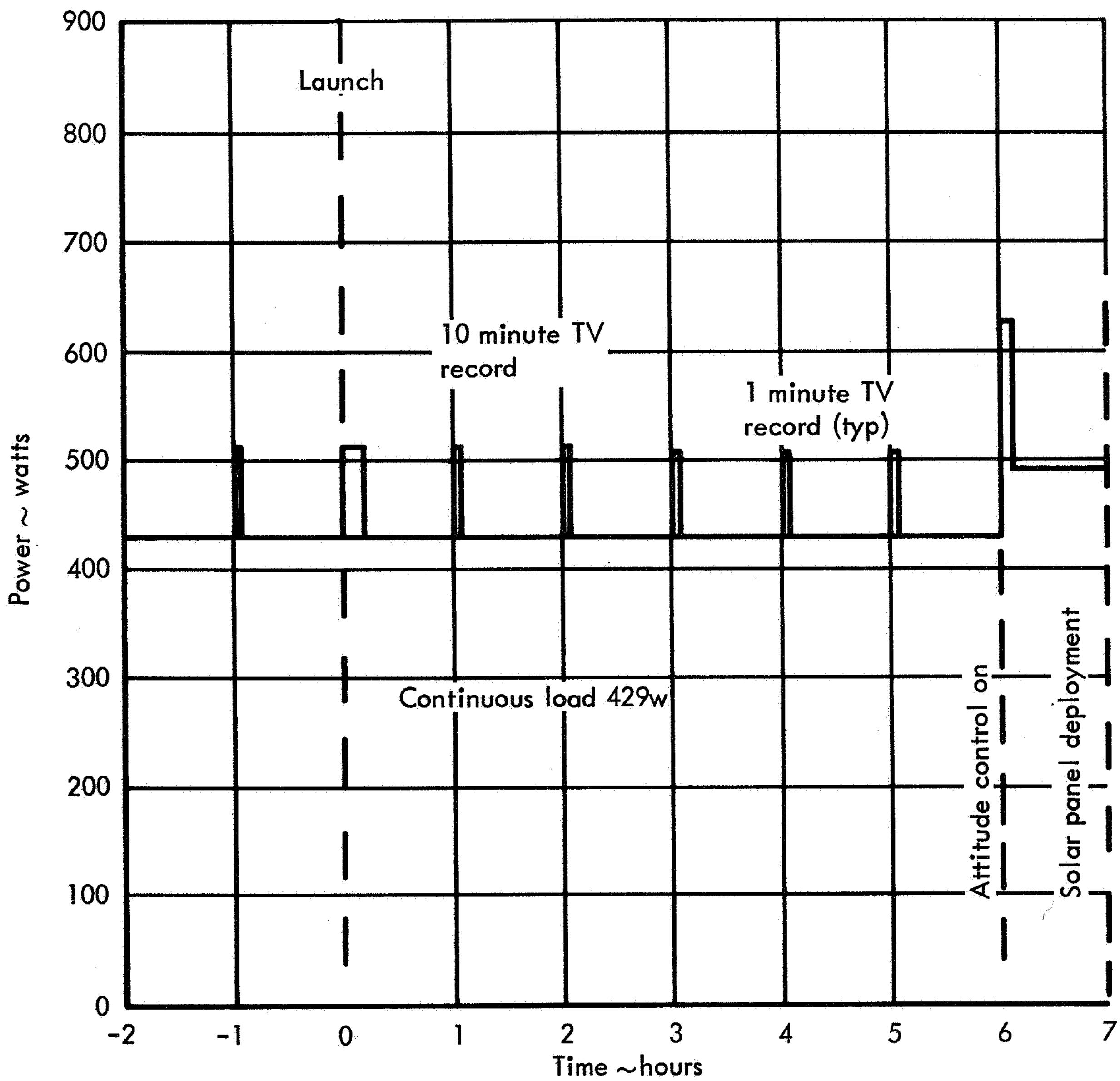


Figure 90. - Prelaunch-launch power profile

and inverters will have associated circuitry for automatic switchover with the capability to override the automatic function through ground command.

The power subsystem will have a minimum in orbit life of one year.

All non-essential failed power using or power conditioning elements of the Spacecraft will be removed from the power bus through appropriate fusing or relay protection.

The switching and conditioning components of the power subsystem must be protected from current transients which could result in gradual degrada-

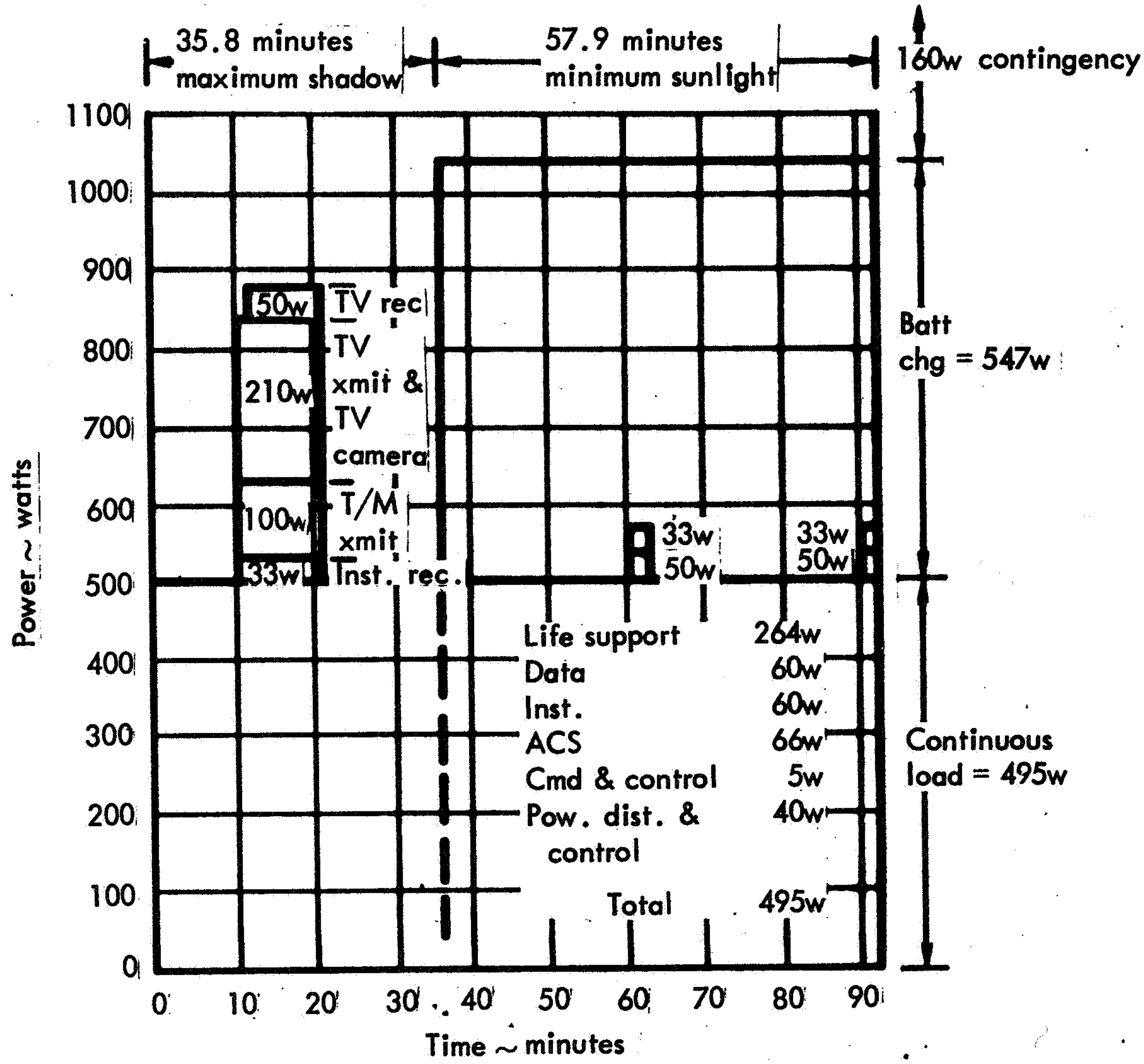


Figure 91. - Power requirements for worst case orbit

tion or failure. Current limiting must be provided at appropriate locations in the power distribution system.

The power distribution system must be designed to minimize cross talk and EMI on all busses. Electromagnetic compatibility, magnetic cleanliness and reliability are prime design constraints.

The basic power sources consisting of solar panels, primary batteries and secondary batteries must be grouped to provide maximum reliability. Implementation of failure mode switching between power sources will be through automatic sensing circuitry. In all instances, the capability to override automatic switching functions by ground command will be provided.

Electrical power and cabling subsystem functional description and performance. - The Primate Spacecraft power subsystem detailed block diagram is shown as figure 92.

TABLE 53. - DETAILED LIST OF PRIMATE SPACECRAFT LOADS (ORBITAL OPERATION)

System	Power-peak	Power-continuous	Type of power
A. Life support and thermal	(watts)	(watts)	
Main fan	350.0	100.0	28 vdc*
Collant fan	30.0	30.0	28 vdc*
O_2 valves	15.0	0	28 vdc*
Miscellaneous valves	20.0	0	28 vdc*
Miscellaneous	26.5	26.5	28 vdc*
UV light	20.0	15.0	115 vac 400 Hz
Fluorescent lights	70.0	60.0	115 vac 400 Hz
Feeder	178.0	0	28 vdc*
Waterer	24.0	0	28 vdc*
Behavioral panel	4.5	4.5	28 vdc*
Behavioral panel elect.	5.0	5.0	28 vdc ± 1%
Thermal	24.0	24.0	
	766.5	264.5	
Moving wall (2)	200.0	0	28 vdc*
Recovery capsule door (2)	120.0	0	28 vdc*
TV door actuator	1.0	0	28 vdc*
	108.7	264.5	
B. Telemetry			
Diplexer	2.0	0	28 vdc*
Data transmitter	100.0	0	28 vdc*
CMD record	3.0	3.0	28 vdc*
Data Record	30.0	30.0	28 vdc*

*Unregulated

TABLE 53. - (Continued)

System	Power-peak	Power-continuous	Type of power
VCO's	10.0	10.0	28 vdc*
Data switch	7.0	7.0	28 vdc*
PCM encoder	10.0	10.0	28 vdc*
TV camera	14.0	0	28 vdc*
TV recorder	50.0	0	28 vdc*
TV transmitter	200.0	0	28 vdc*
	426.0	60.0	
C. Command & control	5.0	5.0	28 vdc*
	5.0	5.0	
D. Attitude control			
Gyros	18.0	10.0	25 vac 3 ϕ 2400 Hz
Pickoff	1.0	.5	28 vac 1ϕ 1600 Hz
Heater	70.0	45.0	28 vdc*
Jet valves	12.0	0	28 vdc*
Sun sensor	1.0	1.0	28 vdc ± 1%
Electronics	10.0	10.0	28 vdc ± 1%
	112.0	66.5	
E. Instrumentation			
No. 1 signal cond.	15.0	15.0	28 vdc ± 1%
No. 2 signal cond.	15.0	15.0	28 vdc ± 1%
Dosimeter	0.5	0.5	28 vdc ± 1%
Data processor	10.0	10.0	28 vdc ± 1%
Biotelemetry receivers	18.0	0	28 vdc*
Biotelemetry discriminator	12.0	0	28 vdc*

*Unregulated

TABLE 53. - concluded

System	Power-peak	Power-continuous	Type of power
Microphone pickups	6.0	0	28 vdc*
Experiment transducers & sensors	15.0	15.0	28 vdc ± 1%
Engineering transducers	51.0	4.5	28 vdc ± 1%
	142.7	60.0	
F. Power conditioning Loss	62.0	39.0	28 vdc*

*Unregulated

The solar array has been subdivided into four 275-watt panels and a body mounted 100-watt panel, each of which feeds the unregulated bus and the battery charger/battery combinations. Battery power will be utilized during that portion of the orbit when the primate spacecraft is eclipsed by the earth for short duration peak loads, and for 1 second during primate feedings. The solar panels provide power to operate the spacecraft and charge the four nickel cadmium batteries during the time the vehicle is in the sunlight. Power sharing between the solar array and batteries can be accomplished with diodes with the operating voltage at the maximum power point of the solar array judiciously chosen and matched to the voltage characteristics of the batteries. The solar array has a growth factor of 160 watts capacity.

If a failure occurs in one of the battery charger/battery combinations, it will be switched off line and the other three batteries will carry the load. The depth of discharge for the remaining batteries will increase to 33% during the dark portion of the orbit.

Failure of two battery charger/battery combinations would increase the depth of discharge to 50%. Most of the mission objectives would be accomplished, but it is doubtful if the one year mission time would be met. The chances of completing the one year mission with two failed batteries would be enhanced if the battery capacities are increased thereby reducing the depth of discharge. Reduced depths of discharge effect an added weight penalty. The increase in battery weight for reduced depths of discharge is shown as figure 93.

Prior to launch, power will be supplied to the spacecraft through the umbilical directly to the main unregulated bus. The ground power operates the spacecraft subsystems and maintains all batteries in a state of full charge.

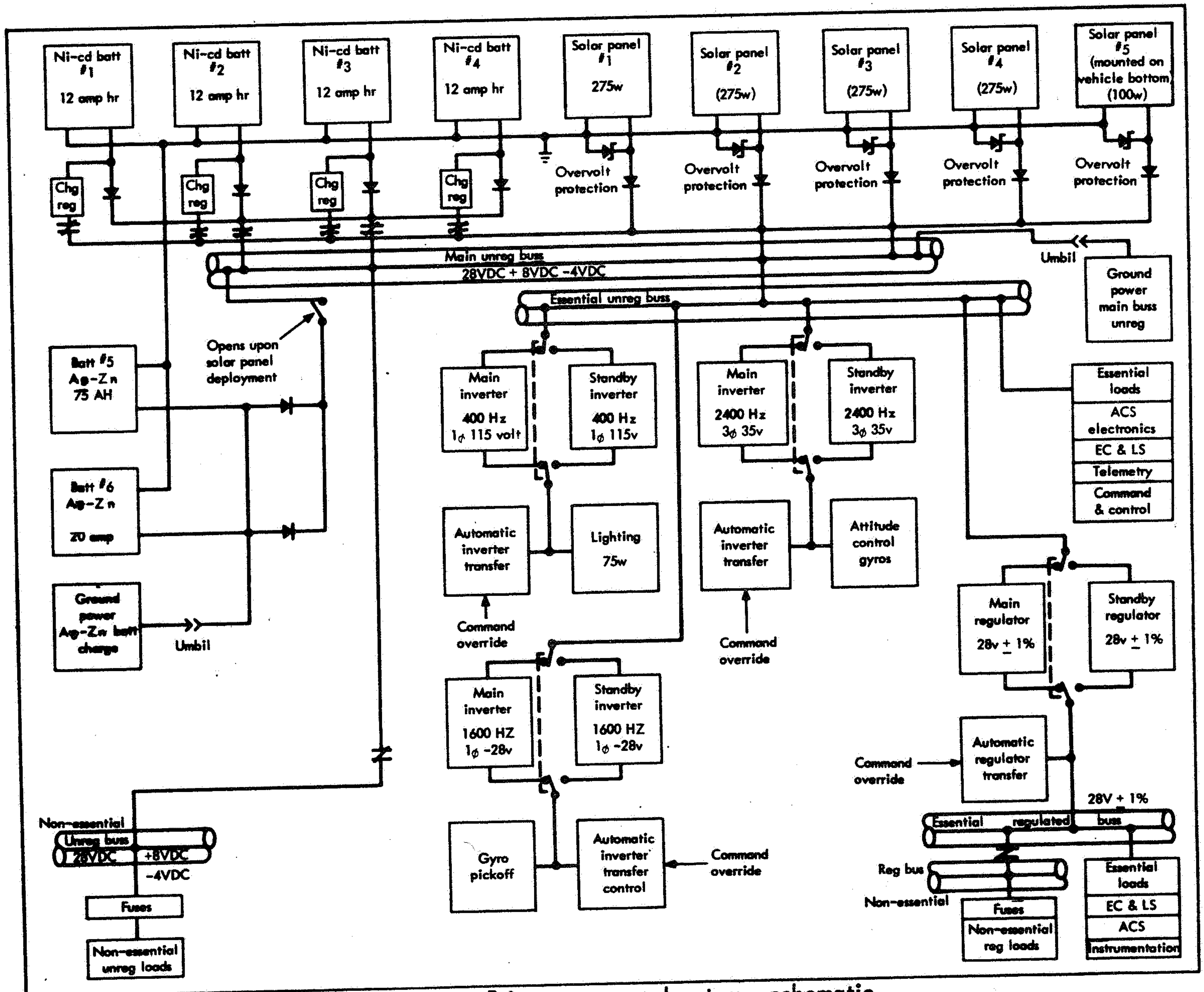


Figure 92. - Primate power subsystem - schematic

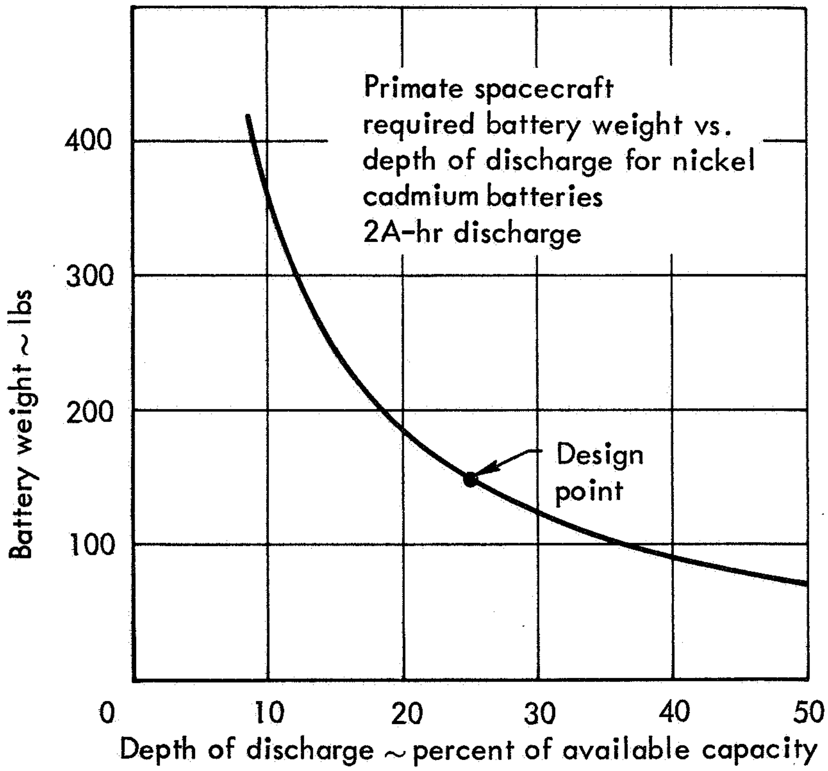


Figure 93. - Primate spacecraft battery weight versus discharge

During the time period from umbilical disconnect to solar panel deployment, power will be supplied by two identical Ag-Zn primary batteries. The deployment of the solar array will cause a switch to open and disconnect the primary batteries from the unregulated bus. Simultaneously, the nickel cadmium batteries are switched to the bus.

The main unregulated bus feeds the essential unregulated bus. The essential unregulated bus supplies dc power to all inverters, and essential subsystems.

Main and standby inverters provide three types of ac power for lighting and attitude control. A failure in the main inverter will be sensed by the automatic inverter transfer circuitry which then switches the standby inverter on-line and switches the main inverter off-line. Similar automatic transfer circuitry is provided for voltage regulator control. Non-essential regulated and unregulated busses can be disconnected from the essential busses by ground command.

Power sources: The Spacecraft will require 32 vdc ground power to be supplied through the umbilical to the unregulated bus. The ground power maintains all batteries at full charge and provides power to operate the spacecraft during prelaunch checkout. The ground power supplies must be capable of delivering at least 1200 watts.

Two silver-zinc primary batteries, each having a capacity of 2250 watt-hours or 75 ampere-hours at a nominal voltage of 30 volts will be used for the prelaunch-launch period internal power source. The primary batteries are connected to the unregulated bus through blocking diodes so that the bus power can be maintained should one primary battery fail.

Failure of a primary battery during launch can be almost completely off-set by automatically switching the nickel-cadmium secondary batteries to the unregulated bus. Normally the secondary batteries would not be connected to the bus until solar panel deployment.

The primary batteries will be manually activated and will be maintained in a temperature controlled environment, 80 + 5 or -10°F both before and after installation in the spacecraft to minimize degradation. Activation will occur shortly before launch.

Silver Zinc primary batteries can withstand several charge discharge cycles under the proper conditions and hence a limited amount of ground operation will be possible. Recharging the primary batteries for use as an emergency power source after the launch phase is also a feasible possibility.

Four 12 ampere-hour nickel cadmium secondary batteries will be connected in parallel to provide a capacity of 48 ampere-hours. Each battery is connected to the unregulated bus through diodes in a manner such that a failure in one of the batteries will not load the bus. The battery voltage will reach 34.0 volts when fully charged. During the dark portion of the orbit, the maximum battery depth of discharge will be approximately 25% and the terminal voltage will vary from 31.5 to 29.1 volts, figure 94.

Associated with each nickel cadmium battery is a charge regulator. The charge regulator adjusts the battery charging current as a function of battery voltage and temperature and prevents excessive gassing within the cells.

In the past, linear regulator circuits have been employed to couple the solar array to the battery. Linear circuits regulate by dissipating excess energy in series or shunt elements and are inherently limited in efficiency; therefore, a switching mode regulator design will be utilized for the Primate Spacecraft charge regulator.

The switching mode regulator will be optimally controlled based upon battery temperature and voltage feedback. An extremal seeking control loop will sense the current input to the battery and based upon voltage and temperature information adjust the charge current to the maximum allowable. This type of battery charge control has been successfully mechanized in the Surveyor spacecraft charge regulator. Under certain conditions, charge regulator efficiencies of 98% have been realized.

The input to the charge regulator is connected to the solar panel bus through a command controlled relay. A failed battery/battery charger combination will be completely removed from the bus.

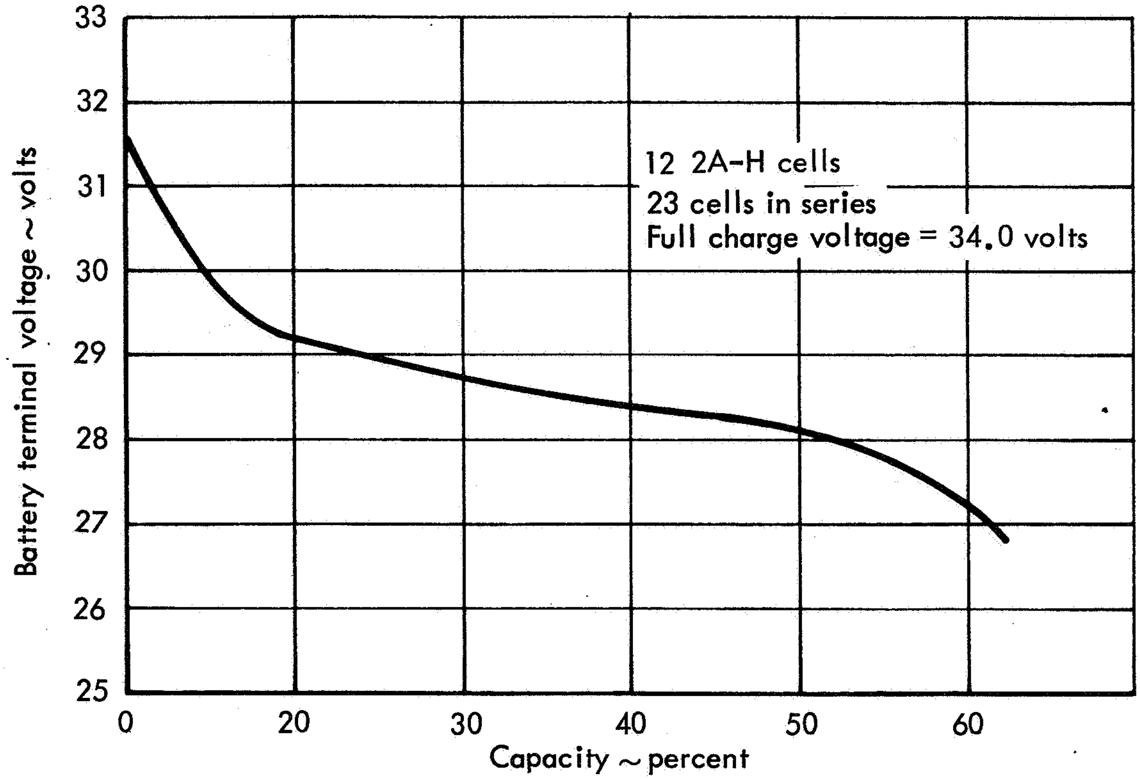


Figure 94. - Nickel-cadmium battery discharge characteristic

The Primate Spacecraft solar array is divided into four, 275-watt panels and one, 100-watt body mounted panel consisting of N/P silicon cells. The panels are subdivided into 48 modules each having an output of approximately 25 watts at 55° C. The modules are connected to the panel bus through blocking diodes and are shunted by zener diodes.

The maximum available solar panel area, consistent with deployment and stowage constraints is approximately 120 square feet. Therefore, an 11-square-foot fixed panel is mounted on the bottom surface of the spacecraft.

Protection circuity: Protection circuitry is required to obviate voltage excursions above or below the design limits on the unregulated bus. The elimination or attenuation of undesirable voltage or current transients which could result in unprogrammed cycling of subsystem functions or which could damage sensitive components is mandatory.

When re-entering the sunlight, the solar array experiences a temporary, approximately 20-minute increase in voltage due to the low temperature of the solar cells. Figure 95 indicates that the array voltage will exceed 60 volts at low temperatures. The maximum power point during the overvoltage condition will be approximately 1900 watts.

One approach to the problem would be to connect the secondary batteries directly to the solar panel bus through relays during the overvoltage period thereby accomplishing battery charging and attenuation of the solar array overvoltage.

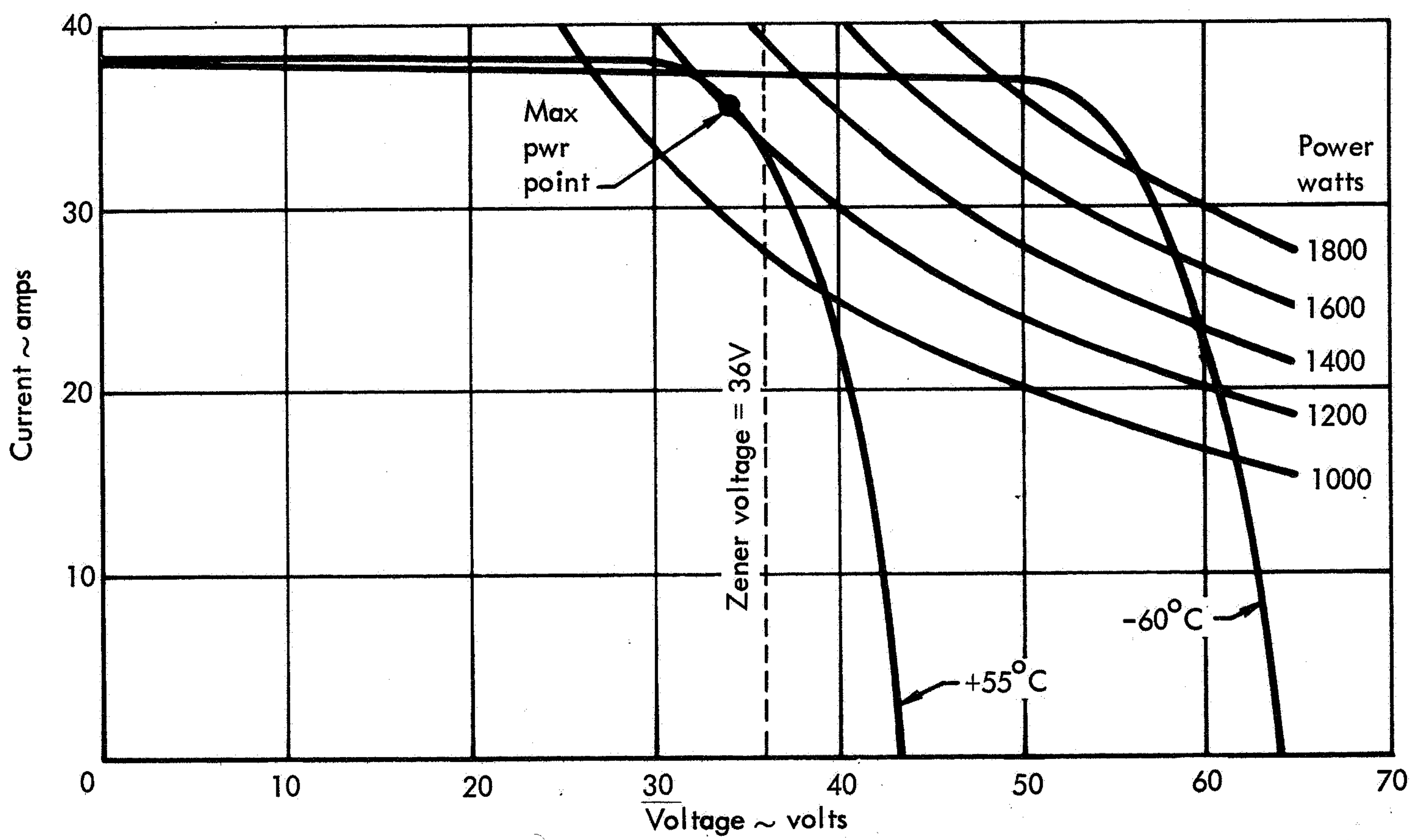


Figure 95. - Primate spacecraft solar array characteristics P/N silicon cells (11.3% EFF)

While this approach effectively reduces the solar panel area, it has the disadvantage of being unreliable. Failure of the relay switching circuitry could result in catastrophic battery failure due to overcharge or failure to charge, and subsequent mission failure.

A simple, reliable approach used on many satellites is to place zener diodes across the output of the solar panel modules as shown in figure 96. This provides loading of the panel when its output exceeds the zener voltage.

If the zener diodes across one of the 25 watt modules fail by open circuiting, the bus voltage would remain within tolerance since the module would be heavily loaded. Should the zener diodes short the associated module would be effectively removed from the bus. The blocking diodes prevent shorted zeners or shorted solar cell modules from loading the bus.

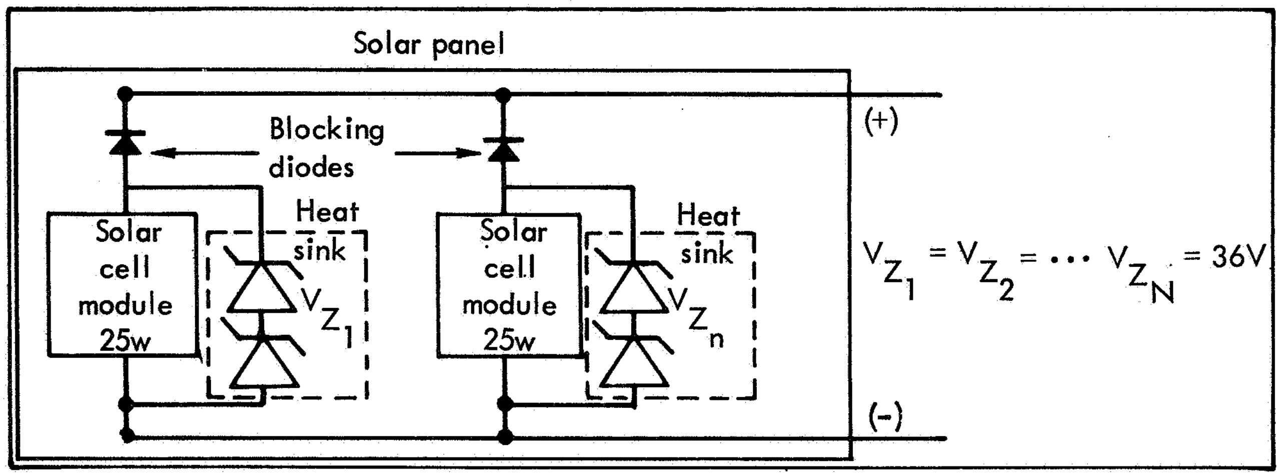


Figure 96. - Zener diode overvoltage protection

Undervoltage protection circuitry will be provided to prevent the battery terminal voltage from falling below 24 volts. The cell voltage of the batteries under this condition would be approximately 1 volt indicating almost complete discharge. Further discharge could result in cell polarity reversal resulting in permanent damage.

The undervoltage circuitry will continuously sense the battery voltage and disconnect all but essential loads from the bus if the batteries reach 25 volts.

The primate spacecraft non-essential loads are fused to protect the power conditioning devices and power sources from faults in these loads. Effective protection is realized only if the capability exists to blow any fuse without damage occurring in the power subsystem.

Littelfuse picofuses are proposed for use in the fuse box. These fuses are presently available in ratings of 0.125 to 5 a. The current vs time to blow characteristic of these fuses indicates that currents up to 35 times greater than the fuse rating can be drawn for short periods of time, figure 97. In order to protect the power conditioning devices and relay contacts, it is imperative that current limiters be incorporated into the power subsystem at vital locations.

The primate spacecraft power subsystem design will utilize current limiters whose limiting values assure that the fuses will blow in the event of a fault but at the same time protect the power carrying relays and conditioning devices.

Regulators: Redundant buck-boost regulators will be utilized in the primate spacecraft. The regulators will be designed to have an output of 28 vdc ±1% over an input voltage range of 24 to 36 vdc. The regulator will be rated at 110 watts continuous output and will maintain the regulated bus at 28 ± 2 volts with a 150% overload for ten seconds. The regulator will be short circuit protected.

The automatic regulator transfer circuitry contains comparators, delay circuitry and relay drivers which logically switch the standby regulator to the regulated bus and the presumed failed regulators to off-line.

A ground command override circuit permits manual regulator switchover in the event that automatic switchover to a previously failed regulator occurs, or if internal failure to switch problems develop. To minimize transients during the regulator transfer time a make-before-break scheme will be utilized and an energy storage network will be placed across the regulated bus output. The sensing circuit will switch regulators if the regulated bus falls and remains outside the limits of 28 ± 2 v.d.c.

Inverters: The Primate Spacecraft requires three types of ac power:

(1) 35 volt, 2400 Hz, three-phase 20 watts

(2) 115 volt, 400 Hz, single phase 75 watts

(3) 28 volt, 1600 Hz, single-phase 1 watt

The inverters are conventional, stepped wave static type inverters. The output waveform is constructed by adding the outputs of several square-wave inverters. The inverters will be short circuit protected.

Each type of ac power is supplied by a redundant inverter pair. The associated automatic transfer circuitry senses the main inverter outputs and switches to the standby inverter in case of failure. The mechanization of the automatic transfer circuitry is similar to that of the regulator transfer circuit.

Distribution: The distribution of power to the using subsystems will be directed by the Command and Control Subsystem. All relays distributing power to vital spacecraft functional elements will have redundant contacts. The power distribution relays will be of the latching type. Power and low level

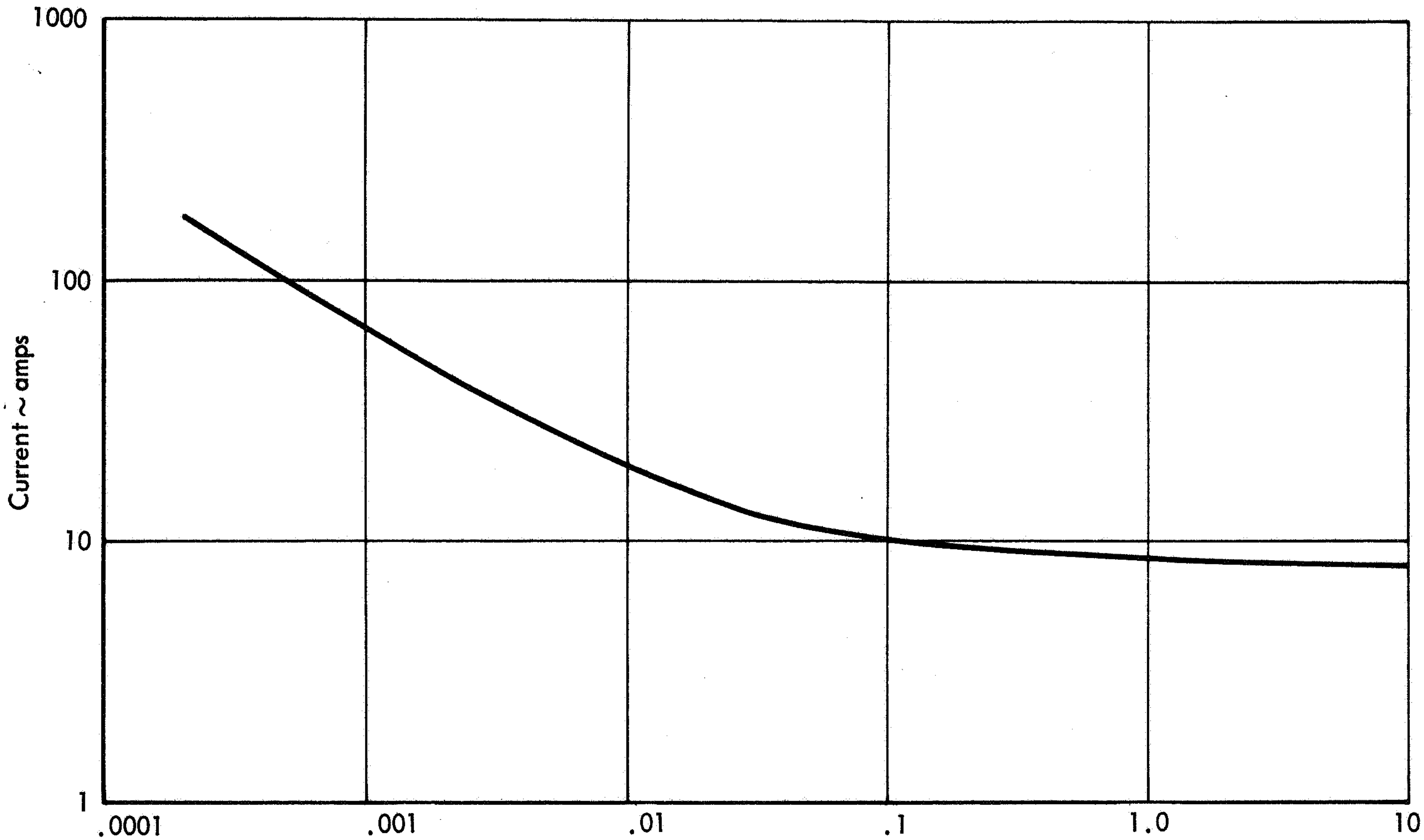


Figure 97. - Fuse current vs. time to blow 5 amp fuse

signals will be carried in separate harnesses wherever possible. The primary harness will be routed around the periphery of the primate spacecraft with smaller harnesses extending radially to the subsystems.

Distribution of power, telemetry, instrumentation and command and control signals is provided by the Primate Spacecraft cabling. The design philosophy will be that a single point ground system be used for the grounding of all return paths. Extensive shielding along with adequate bonding and cable harnessing techniques will be used to insure the suppression of electromagnetic interference.

Signal return paths will be routed with the signal current path as shielded twisted pairs. Case ground will not be used as a signal return path. Non-RF shields will be commoned at only one end while shields of RF cables will be connected to chassis at both ends and rf coaxial cable will be routed as closely as is possible to chassis and structure. To minimize cross coupling and ground noise the power returns will be commoned at the power source negative bus which will be located near the power sources and will be connected to the single point ground. The spacecraft frame ground will incorporate electrical bonding where necessary to bridge mechanical discontinuities and to assure that all cases will be at nearly the same potential.

The distribution of d.c. and single phase a.c. will be based on a two wire system via twisted-shielded pairs. The three-phase a.c. power will be distributed via a four wire system. Power and signal cabling will be routed separately wherever possible.

The cabling will be designed and fabricated to be compatible with the requirements of the life support system with regard to contaminants. The cabling will function with a high degree of reliability when exposed to the environment of near earth space, especially temperature, radiation and pressure extremes. In general the wiring will be in accordance with MIL-W-16878D.

Pyrotechnic circuitry: The Primate Spacecraft pyrotechnic block diagram figure 98, indicates a pyrotechnic control unit PCU, a nickel-cadmium battery and the pyrotechnic actuators. Housed within the PCU is the automatic sequencer and the pyro firing circuits.

The PCU will be designed to prevent any single or common failure mode from both arming and commanding a hazardous pyrotechnic event and will conform to Range Safety Manual AFETRM 127-1.

The primate spacecraft PCU requires two inputs from the Command Module:

(1) Separate spacecraft from rack command.

(2) Deploy spacecraft solar panel command.

The spacecraft will be separated from the rack by activating the explosive devices which attach the spacecraft to the rack.

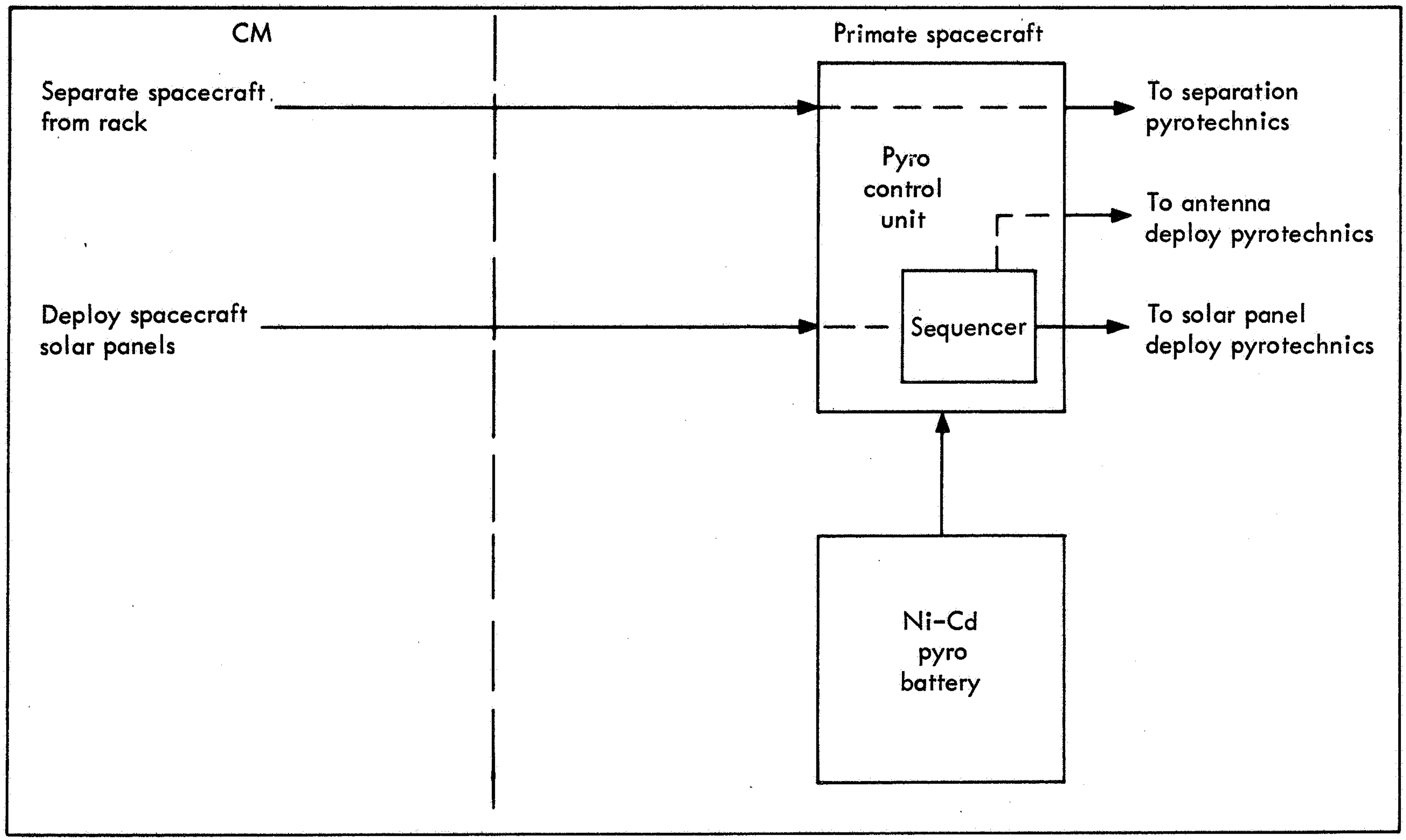


Figure 98 - Primate spacecraft pyrotechnic circuitry

The primate spacecraft solar panels will be deployed on command from the Command Module. The deployment of the solar array will start a timer and sequencer which will deploy the spacecraft antennas at pre-selected time intervals.

All pyrotechnics will be designed to maximize the containment of debris or contaminants.

The Primate Spacecraft pyrotechnic wiring will consist of Northrop developed EMI shielded cabling which is superior to standard copper braid shielding at both low and high frequencies.

EMI shielding for wiring and inter-connecting cables consists of two .fferent magnetic materials in a braid configuration. The outer braid conists of a low permeability, high satuaration material, and the inner braid :onsists of a high permeability material that saturates at low levels. As indicated in figure 99, the outer braid provides moderate shielding at low frequencies and good shielding at high frequencies. The inner braid, on the other hand, furnishes relatively little shielding at high frequencies, but provides excellent shielding at low frequencies.

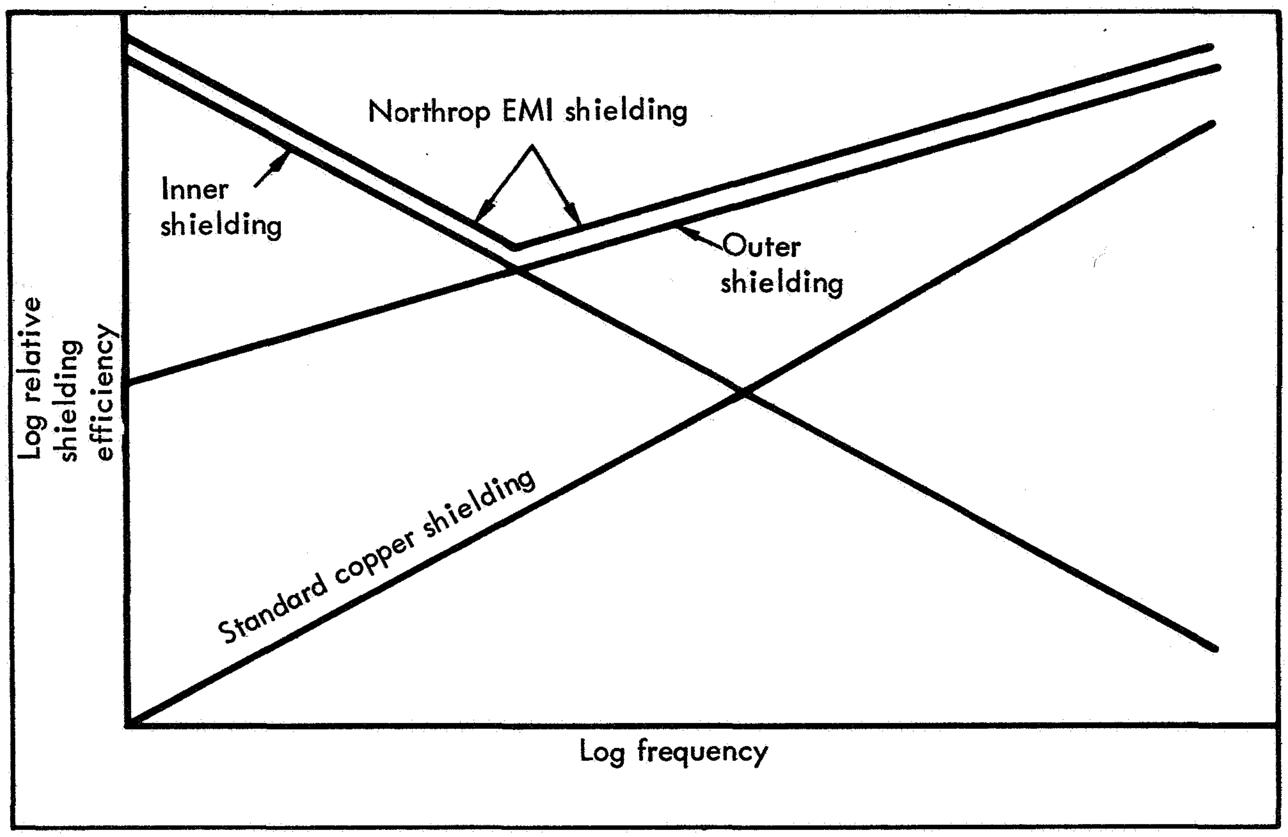


Figure 99. - Shielding efficiency

Use of high permeability shielding (tape, foil, braid, etc.) is not new in shielding from magnetic fields. However, this type of shielding becomes saturated in a strong magnetic field, at both low and high frequencies. This can result in the energy of the magnetic field passing through the high permeability shielding materials, since the energy is greater than the absorption and reflection loss capability of the shielding material. The new EMI shielding overcomes this weakness due to the capability of the low permeability, high saturation material to provide the necessary protection to the high permeability material that saturates at low intensity levels. The outer braid shielding reduces high intensity magnetic field propagated noise to a low level so that the inner braid shielding can function at peak efficiency, thus bypassing the interfering magnetic field around the sensitive conductor and conducting the balance of the propagated noise (electrical field) to ground. The end result is a shielding combination that is effective throughout a broad frequency range.

The preferred grounding method for the new EMI shielding is by the "multiple ground point" or "structure return" method. Interface cables shielded with the new shielding must be terminated at both ends and at every connector to the ground system (rather than at one end, or carrying the shield through the connector).

Advance development areas. - All elements of the Primate Spacecraft power subsystem are within the state-of-the-art capability and no developmental programs are required.

Preliminary equipment list. - The initial estimate of equipment requirements is summarized in table 54.

Attitude Control

This section presents the analytic design of the mass expulsion Attitude Control Subsystem selected for the Primate Spacecraft. The selected approach is the result of an extensive trade-study (ref. 6). The subsystem is further characterized by the primary utilization of three gas bearing integrating rate gyros which both establish and maintain a three-axis reference as well as derive and provide complete vehicle attitude information.

The degree of stability and attitude control which is achievable by any spacecraft can be determined, with a high degree of certainty, only by considering the effects of potential disturbances. Consequently, the nature and effects of both externally and internally generated disturbances were evaluated.

The resulting subsystem is capable of satisfying all of the mission requirements and imposed constraints as well as facilitating the Apollo docking procedure (by means of a maneuvering capability) for such purposes as primate retrieval. This is all accomplished with a relatively conventional design, which because of its inherent simplicity, should possess high reliability.

TABLE 54. - ELECTRICAL POWER AND CABLING PRELIMINARY EQUIPMENT LIST

Item No.	Description	Suggested Manufacturer	Part No.	Quantity per Spacecraft*
1.	12 Amp-Hr. 22 cell Ni-Cd Battery	Gulton Inc.	VO-12-HS Cells	4
2.	2250 Wt. Hr. Ag-Zn Battery	Eagle-Picher	#22585 Cells	2
3.	Solar Cells N/P Silicon 1 x 2 cm	Hoffman or Heliotek		52,000
4.	Charge Regulator	N.S.L.		4
5.	Inverter 400 Hz, 1Ø, 115V, 80W	Engineered Magnetics		2
6.	Inverter 2400 Hz, 3Ø, 35V, 20W	Engineered Magnetics		2
7.	Inverter 1600 Hz, 1Ø, 28V 1W	Engineered Magnetics		2
8.	Power Switching and Logic Module	N.S.L		1
9.	Harness (cabling)	N.S.L		1
10.	Pyrotechnic Control Unit	N.S.L		1
11.	Pyro Battery	Sonotone		1

Attitude control subsystem requirements. - The requirements placed upon the attitude control subsystem originate directly from the mission objectives and indirectly from other subsystems such as electric power and thermal control.

Additionally constraining factors also arise from the selected orbit, the size and shape of the spacecraft, and the nature of the mission, i.e., the experiments.

These considerations have served to establish the following attitude control design requirements and constraints:

(1) Requirements

(a) Life: One year design point

(b) Orientation: Sun pointing

(c) Attitude accuracy: ± 10 degrees (L.O.S. to sun)

(d) Primate retrieval: Facilitate Apollo docking

(2) Constraints

(a) Orbit: 250 n.mi., 28.5 degrees inclination

(b) Experiment: 0.001 g maximum continuous acceleration

0.01 g maximum transient acceleration

ACS shutdown capability

(c) Vehicle mass properties: Knowledge of the vehicle mass properties is essential to the design of the ACS. The rotational dynamics of a body are directly related to the moments of inertia and the location of the center of gravity of the vehicle.

The composite vehicle and its subsystems were evaluated with regard to their mass properties. Such factors as the main structure, sub-structures, solar array, subsystem locations and life support (expendables) equipment were evaluated. The results of this investigation are summarized in figure 100.

Attitude control subsystem description and performance. - The selected design of the Attitude Control Subsystem is essentially that of a conventional bang-bang type with rate gyro damping. The design employs a dual 3-axis reaction jet system, and a dual 2-axis sun sensor system. The two reaction jet systems, each consisting of 6 jets, while physically independent, normally operate simultaneously under common control authority. The 3-axis reference system consists of gas bearing integrating rate gyros operating in a typical strap-down mode to supply both attitude position and attitude rate information in pitch and yaw, and just attitude rate in roll.

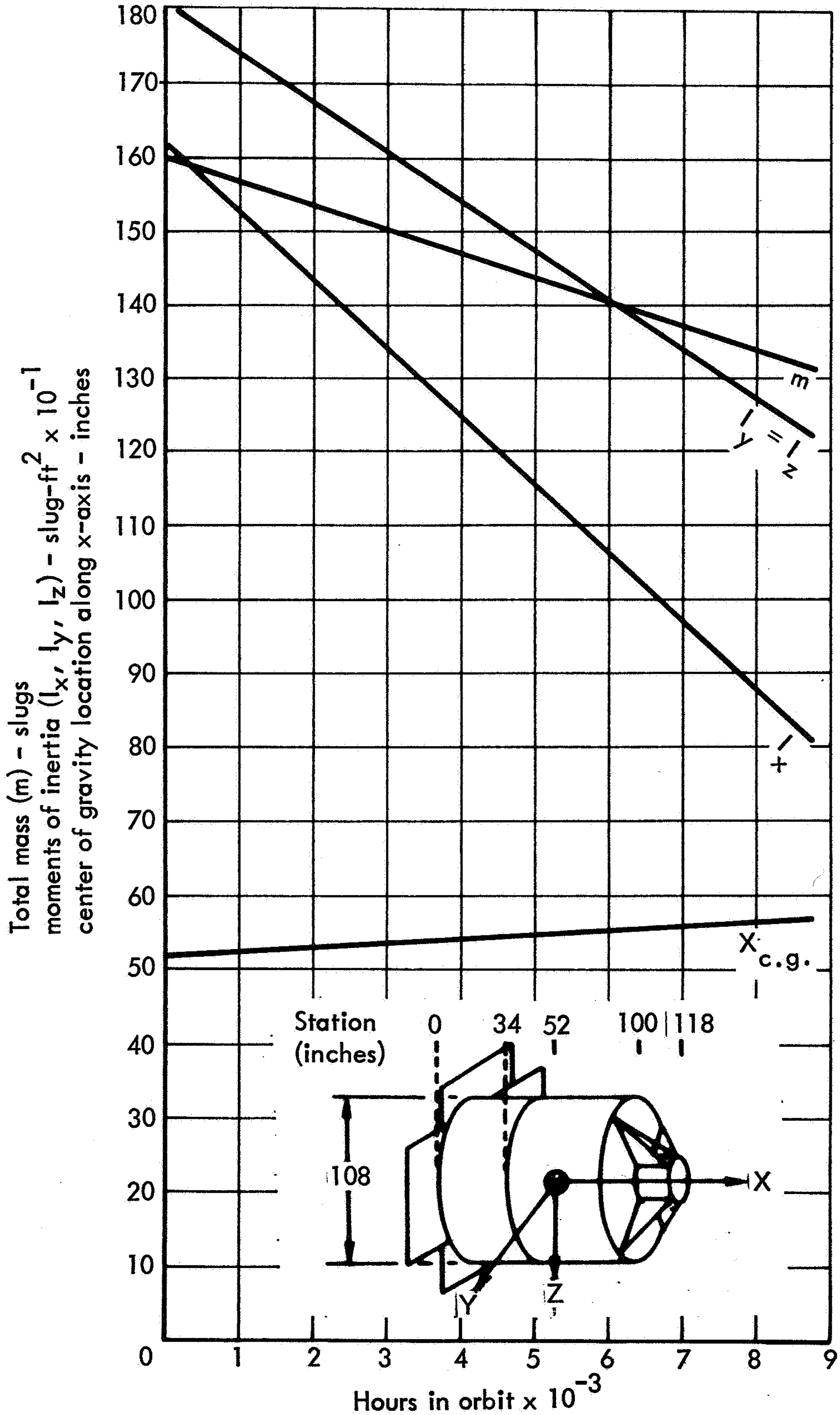


Figure 100. - Primate spacecraft module mass properties history

The Attitude Control Subsystem may, upon command, be deactivated and reactivated for accommodating either experiments or docking. In addition, the capability is provided for maneuvering the spacecraft to any desired attitude. A block diagram of the basic Attitude Control Subsystem design is shown in figure 101.

Sun sensors: The Attitude Control Subsystem employs a dual 2-axis sun sensor system. The two systems, each consisting of 4 sensors, are so arranged on both ends of the spacecraft such that together they encompass a solid spherical detection angle. The sun sensor geometry is depicted in figure 102. Their implementation is devised so that the group-of-four coincident with the solar array end of the spacecraft generates a minimum attitude error signal while the group-of-four on the docking structure end of the spacecraft generate a maximum attitude error signal whenever the respective ends of the vehicle face the sun.

The primary function of the sun sensor system is for acquiring and up-dating the sun reference. It can be observed, from the discussion above and figure 102, that the implemented logic will normally tend to orient the solar array end of the spacecraft towards the sun. The capability is provided for reversing the above mentioned logic such that the docking structure end of the spacecraft would tend to face the sun, and by adding an appropriate command into the control loop, any desired attitude between the two extremes can be attained. This procedure will greatly facilitate Apollo docking because the spacecraft can be oriented to an optimum attitude with respect to the sun in order to provide good visibility and definition. Once the sun is acquired, the 3-axis gyro reference system will maintain the established reference throughout the orbit.

The sun sensor mechanical assembly is completely passive, incorporating no moving parts, and consequently has high inherent reliability.

Gyros: The 3-axis reference system consists of three floated rate integrating gyros which are capable of providing an exceptionally high-level of performance. The rate integrating gyro is so named because the displacement of its precession axis (gimbal) is proportional to the integral of the angular rate input. However, the precession angle (gimbal freedom) capability of even the wide-angle gyros of this type are usually limited to angles smaller than 10 degrees, by employing spring stops. This is done in order to reduce gyro cross coupling errors. Therefore, whenever an application would normally result in relatively large gimbal angles, something must be done to prevent bottoming against the stops. The only thing that can be done is to close the loop around the gyro (electrical caging) which in turn makes the gyro output proportional to angular rate rather than angular position. At first, it may seem that this mode of operation eliminates the gyro's usefulness as an attitude reference source, but it actually turns out to be a blessing in disguise. This mode of operation is selected because a rate gyro affords much better damping than derived rate schemes, providing that the gyro's threshold is below the desired limit cycle rate. Therein lies the shortcoming of most typical spring restrained rate gyros, but a rate integrating gyro operating in the rate mode is capable of much lower input threshould sensitivity, wider dynamic range and greater accuracy than is possible with the conventional rate gyro.

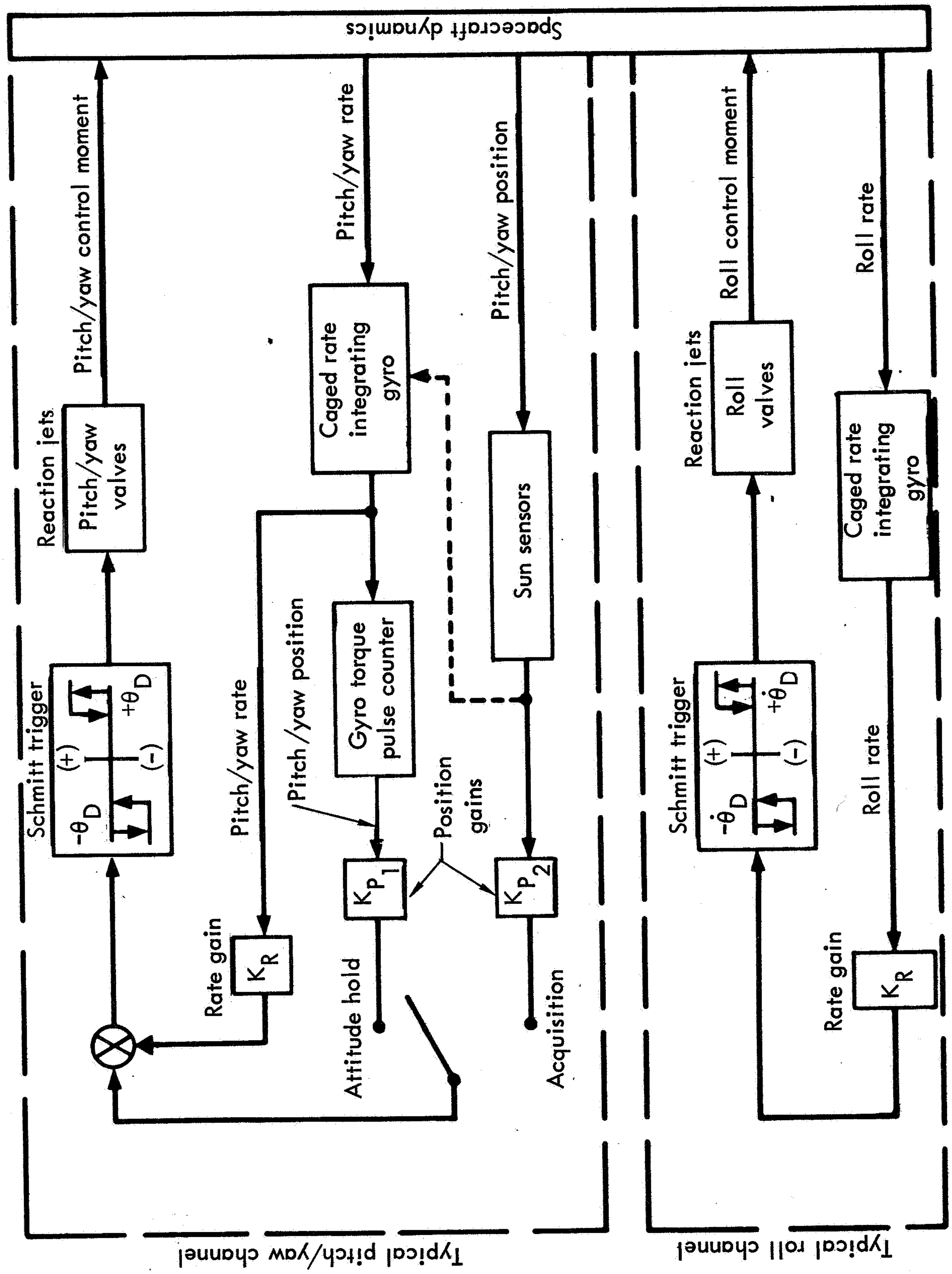


Figure 101. - Attitude control system - block diagram

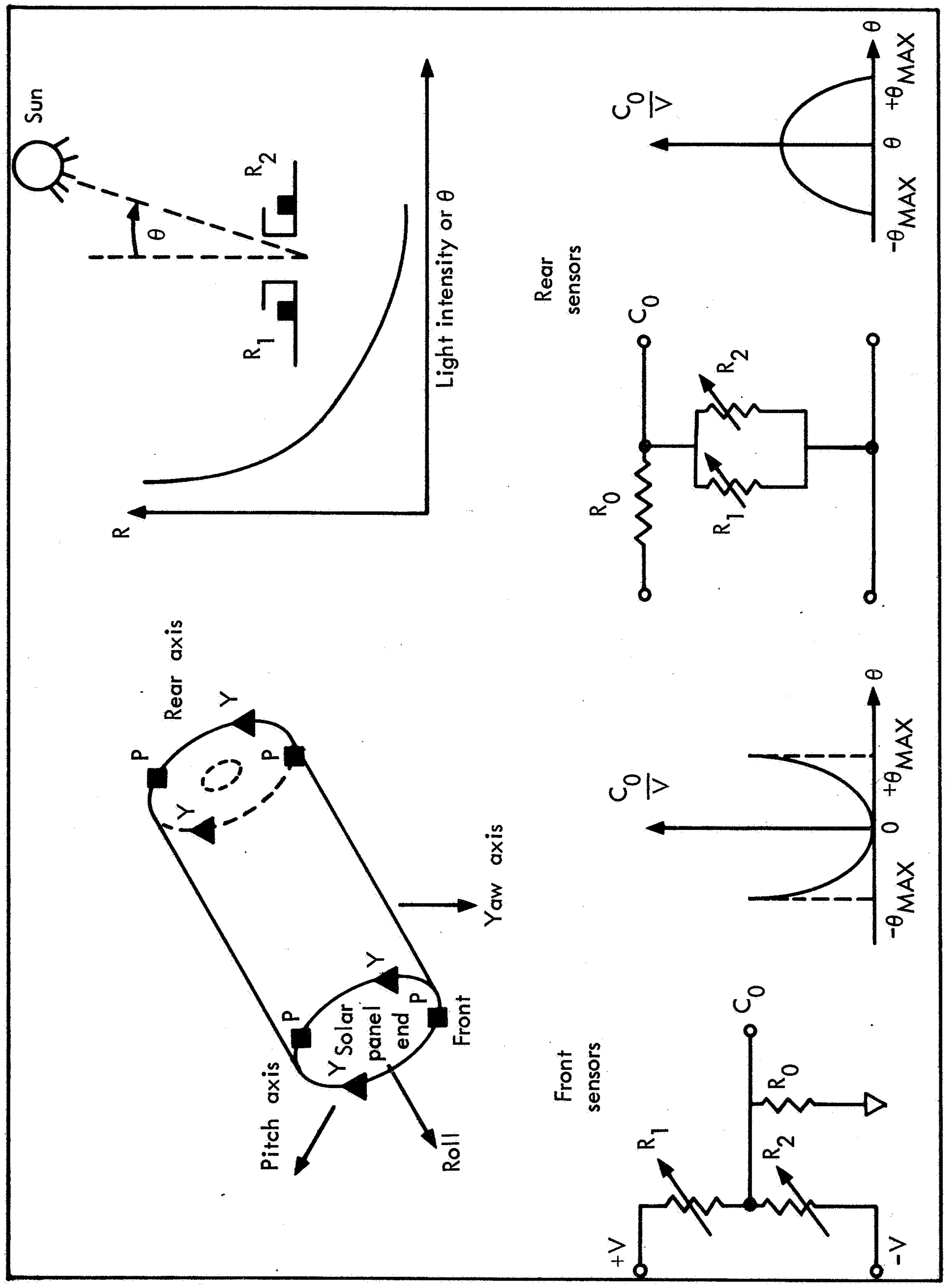


Figure 102. - Sun sensor geometry

It is still necessary, however, to provide for pitch and yaw attitude position information since the sun sensors are dormant whenever the sun is occulted by the earth. This is accomplished by modifying the electrical cage into a pulsed torque rebalance loop. The cage or rebalance loop is closed through the gyro torquers by use of discrete incremental current pulses which are regulated in both width and amplitude. Each pulse represents a precise incremental angle, and the sum of pulses required to cage the gyro (maintain it as its null) represents the total angular error. A block diagram of this gyro configuration is shown in figure 103.

A statement of justification is in order regarding the use of gyros on a long life vehicle. The operational life of even the finest of floated rate integrating gyros with conventional ball spin bearings does not usually exceed 3000 hours, a fact which often promotes apprehension regarding their long term reliability. The gyros utilized in this design are of the floated rate integrating type, but the rotor spins on hydrodynamic gas bearings, which do not require a supply of pressurized gas as opposed to the hydrostatic type which does. The great inherent advantage here is total absence of wear as contrasted with conventional ball bearings which are most often the limiting factors in gyro life. The limiting life factor of a hydrodynamic gas bearing gyro is the number of starts to which it is subjected. The operational life of the gas bearing floated gyro is far in excess of 10,000 hours when the unit is used in continuous operation.

Control unit: The concept of a control unit in a mass expulsion attitude control system usually implies the switching (or triggering) design technique.

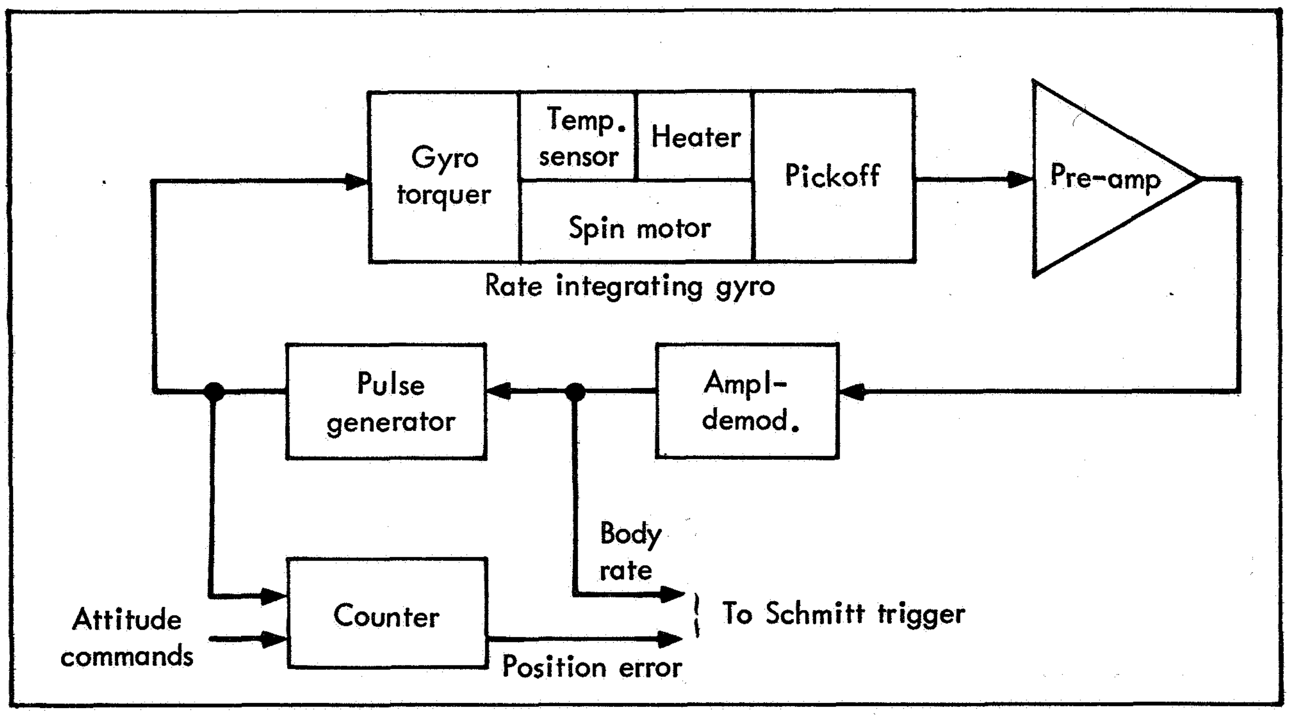


Figure 103. - Gyro configuration (pitch & yaw)

Since the advent of the on-off (bang-bang) attitude controller, switching techniques have evolved which are more sophisticated than the original. They generally all offer some degree of improvement over the conventional approach. This attitude control subsystem design is based on the conventional bang-bang approach which utilizes a Schmitt trigger switching circuit. Depending on the level of the input signal, the circuit will reside in either of two stable states, with the switching action occuring at some specified threshold level. Thus, the circuit also functions as an amplitude comparator indicating by one or the other of its stable states whether the input voltage is above or below the threshold. If the loop gain of this circuit is greater than unity the switch displays a form of hysteresis, switching off at a different voltage level from that when it is switching on. This is a desirable characteristic for this application. The basic characteristics of the trigger are shown in figure 104. It has a deadzone of $\pm\theta_2$ and a per unit hysteresis h. Switch-on occurs at $\pm\theta_2$ and switch-off at $\pm\theta_1$. After switching on, a signal is delivered to the jet valve select logic commanding an appropriate torque. Application of this torque continues until after the trigger switches off.

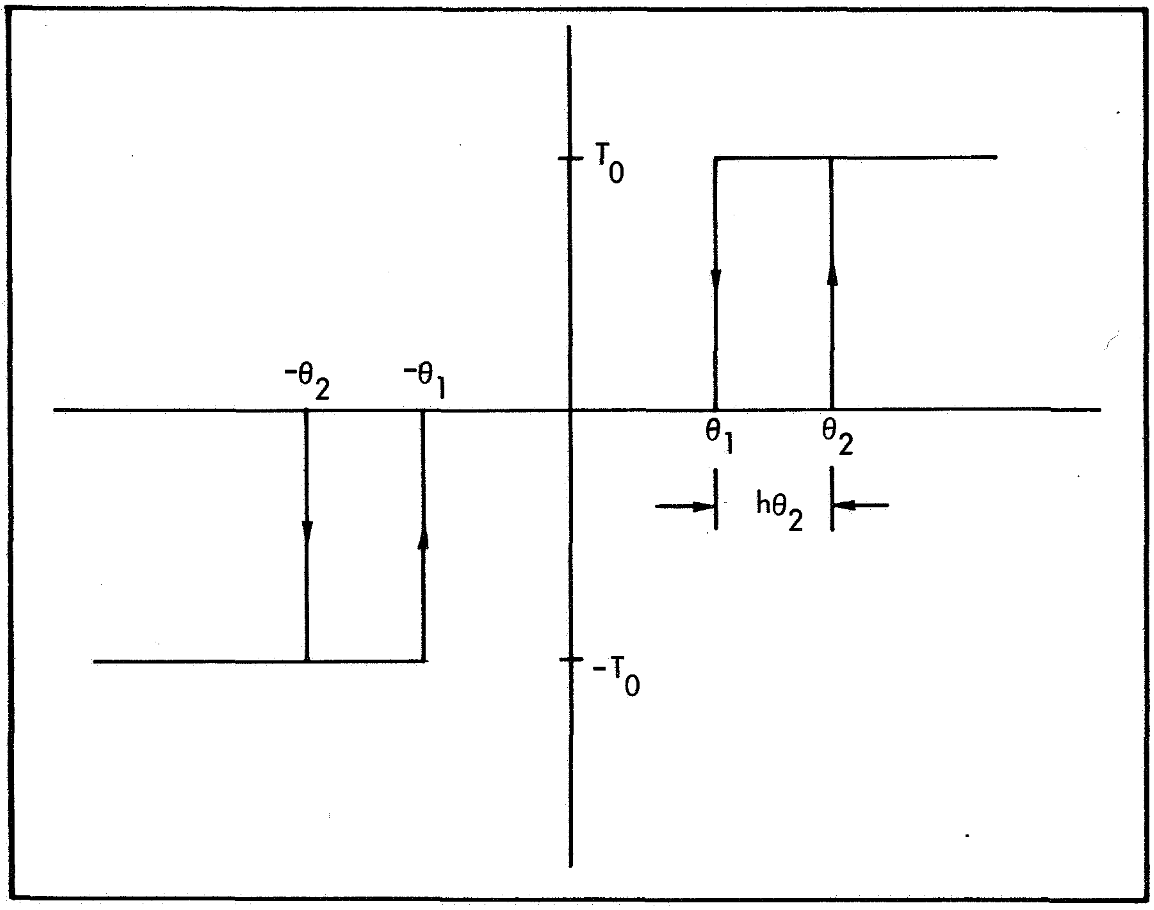


Figure 104. - Switch characteristics

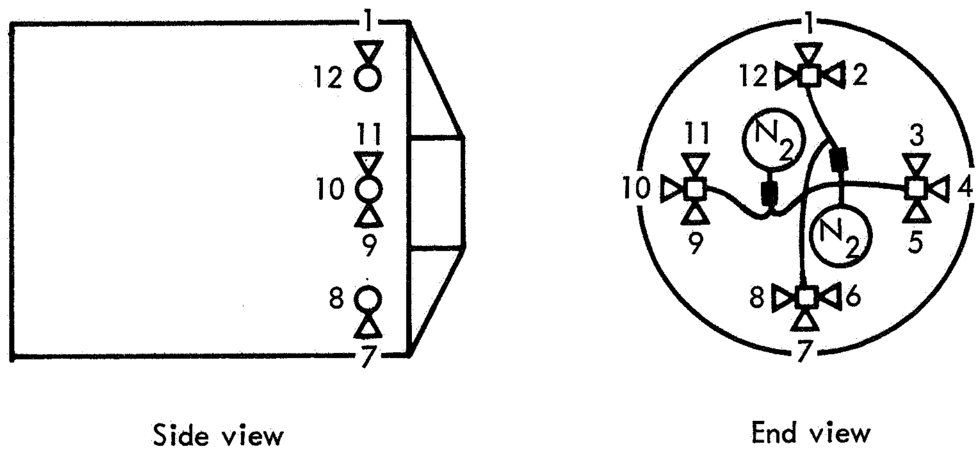
Reaction jets: Low thrust reaction jets are extremely suitable for satisfying the spacecraft's attitude control requirements. Cold gas systems, especially stored compressed nitrogen, have received the most extensive use and application in spacecraft. They are very simple in concept and their principles of operation are straight-forward in theory. Of the cold gases, nitrogen results in one of the heaviest systems; however, its inherent system simplicity and reliability, resulting from proven state-of-the-art techniques in contamination control and no-leakage sealing methods, make it the uncontended choice for this application. Only nitrogen cold gas systems have demonstrated the operational capability for long duration space missions.

The selected reaction jet configuration is essentially a dual 3-axis propulsion system. Twelve jets are utilized in four clusters, each consisting of three jets. There are two propellant tanks, each of which feed two opposing jet clusters. This reaction jet geometry is depicted in figure 105. The two jet systems, which are physically independent, normally operate simultaneously under common control authority. One of the principal features of this configuration is that a valve-open situation (the worst conceivable failure) does not abort the mission. This is accomplished by carrying sufficient propellant in each of the dual tanks to both maintain control and aid in countering the open-valve failure. In addition, the selected configuration requires a minimum of operational logic complexity, and its cumulative effect on orbital velocity is negligible.

Attitude control system performance. - With the spacecraft in its designated orbit and operating, the sun sensors then point the roll axis at the sun in the manner previously described. Roll gyro operation is required primarily to preclude excessive radial accelerations. This is accomplished by measuring roll rate, and in conjunction with a rate threshold detector, limits these rates to the threshold values. Pitch and yaw attitude are generated by the sun sensors while rate damping about these axes are generated by the respective gyros. Sun lock-on is achieved when the roll axis (solar panel end) points toward the sun to within $\pm$ 10 degrees. This terminates the acquisition mode and activates the attitude-hold mode.

In the attitude-hold mode, pitch and yaw attitude and rates are sensed and generated by the respective gyros working in conjunction with "integrating" circuits. The sun sensors are active in this mode, and their sole function is to provide a reference up-date if required.

The Attitude Control Subsystem performance capabilities and specifications are summarized in table 55.



Firing Matrix	
Rotation:	Jets
+ Roll:	2, 8, 5, 11
- Roll:	6, 12, 3, 9
+ Yaw:	2, 6, 4
- Yaw:	8, 12, 10
+ Pitch:	7, 5, 9
- Pitch:	1, 3, 11

0.5 Lb thrust jets: 1, 4, 7, 10
0.25 Lb thrust jets: (all others)

Figure 105. - Reaction-jet geometry

TABLE 55. - ATTITUDE CONTROL SUBSYSTEM PERFORMANCE CAPABILITIES AND SPECIFICATIONS

Pitch/yaw control:	Attitude hold (limit cycle)	
	Attitude threshold	= ±10 degrees
	Average rate	= ±0.002 degrees/second
	Average period	= 5.5 hours
Roll control:	Attitude rate limited	
	Rate threshold	= ±0.2 degrees/second
	Average rate	= ±0.002 degrees/second
Control moment:	±4 Pound-feet (all axes)	
Reaction jet size:	0.5 Pound (radial jets); 0.25 Pound (tangential jets)	
Propellant:	Stored compressed nitrogen gas	
	Weight of gas	= 40 pounds
	Weight of gas & bottles	= 60 pounds
	Size of bottles (2)	= 10 inches OD
Gyros (type):	Rate integrating with torque rebalance	
	Floated gimbals	
	Gas spin bearings	
Sun sensors (FOV):	4π Steradians	
Power consumption:	90 watts (peak), 70 watts (average)	
Total ACS weight:	150 Pounds (including (propellant)	

Limit cycle characteristics: The limit cycle characteristics are set forth below. The undisturbed single-axis rotational motion of the spacecraft about its center of gravity is represented by the equation:

$$I\ddot{\theta} = M_C$$

where I is the moment of inertia,

$\ddot{\theta}$ is the angular acceleration, and

M_C is the applied control moment.

The angular acceleration can be written in the form:

$$\ddot{\theta} = \frac{d\dot{\theta}}{dt} \approx \frac{\Delta\dot{\theta}}{\Delta t}$$

where $\Delta\dot{\theta}$ is the change in angular velocity

Δt is the duration of torque application

therefore:

$$\Delta\dot{\theta} = \frac{M_C \Delta t}{I}$$

For a symmetrical limit cycle, which yields the worst gas consumption, the limit cycle rate $\dot{\theta}_{LC}$ is equal to one-half of the change in angular velocity produced as a result of torque generation. This equation can then be expressed in terms of symmetrical limit cycle rate as:

$$\dot{\theta}_{LC} = \frac{M_C \Delta t}{2I}$$

where the product $M_C \Delta t$ represents the torque impulse which is equivalent to the magnitude of the generated angular momentum. The conventional signle pulse limit cycle is optimized (minimum limit cycle rates) if the duration of effective torque application is minimized. This implies a realistic minimum pulse width capability of the reaction jets taking into consideration both valve dynamics and delivery of the steady-state specific impulse from the propellant. Considering these factors, a conservative value for the effective minimum pulse width was selected to be equivalent to 20 milliseconds. Considering the moment of inertia characteristics of the vehicle, as presented in figure 102, a plot is presented in figure 106 of the achievable limit cycle rate as a function of moment of inertia for various values of generated control moment.

The torque impulse, $M_C \Delta t$, has been shown to be equivalent to the magnitude of the generated angular momentum. The magnitude of the generated angular momentum in the limit cycle is of significance to the sizing of the system. Figure 107 is a plot of this parameter as a function of moment of inertia for various values of limit cycle rate.

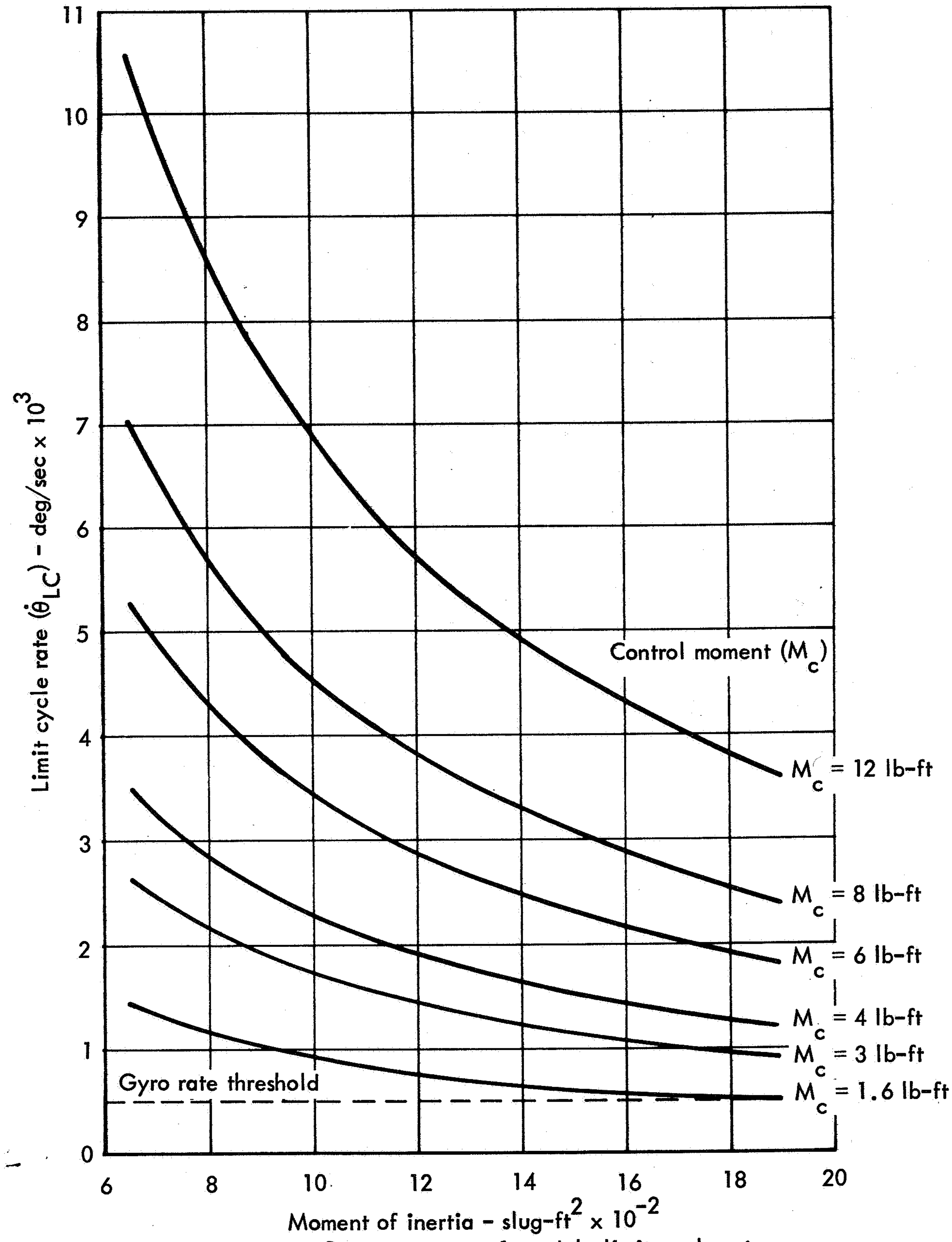


Figure 106. - Primate spacecraft module limit cycle rate

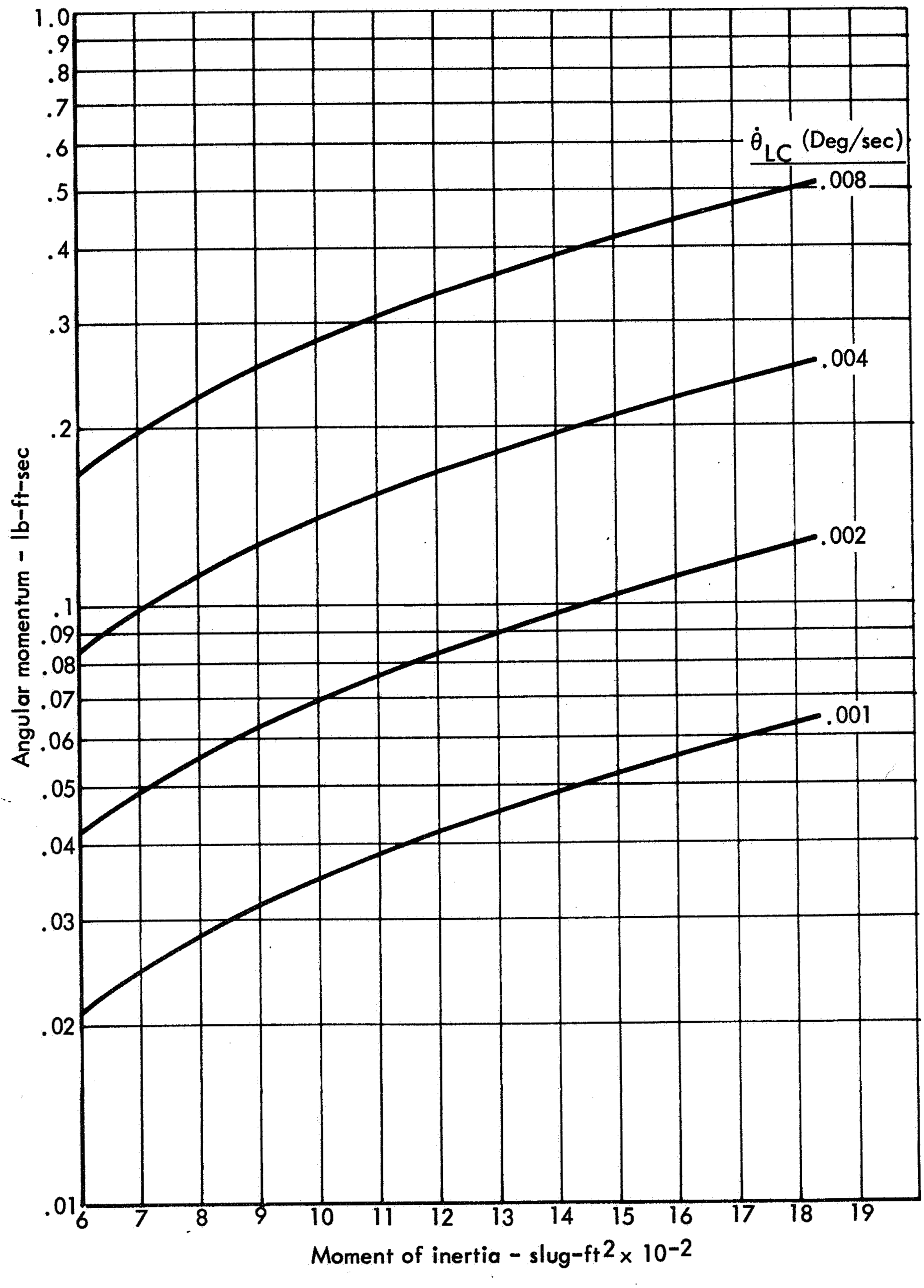


Figure 107. - Primate spacecraft module angular momentum in the limit cycle (per axis)

A plot of control moment as a function of angular momentum for various values of jet on-time, $t_{on} \approx t$, is presented in figure 108. Since the average control moment-lever arm about all vehicle axes is four feet, it is possible to cross plot the required control thrust on the same plot.

The purpose of selecting a minimum jet pulse width was to minimize the limit cycle rate, and the reason for minimizing the limit cycle rate is because it is the most critical parameter in determining propellant weight. The amount of gas consumed per year (GCPY) due to limit cycle operation about the pitch or yaw axis (it will be shown that the equivalent roll axis value is much lower) can be expressed as:

$$GCPY = 5.5 \times 10^5 \frac{\dot{\theta}_{LC}{}^2 I}{\theta_{LC} \ell isp} \quad lb$$

Where θ_{LC} is the limit cycle angle (limit cycle angle for pitch and yaw; no regular limit cycle occurs in roll) ℓ is the average control moment-lever arm (lever arm changes slightly throughout mission because of shifting c.g.), isp is the effective specific impulse, total impulse per unit weight of tank and propellant. The limit cycle angle will be ten degrees. The average control moment-lever arm is four feet. The effective specific impulse of nitrogen for the operating pulse width range is 31.4 seconds measured at 300° R and 3500 psia. Substituting the foregoing values into the previous equation for the appropriate parameters results in the following expression for gas consumption per year:

$$GCPY = 0.05 \; I\dot{\theta}_{LC}$$

The equation is plotted in figure 109.

External disturbances: The spacecraft will be acted upon by environmental forces resulting in disturbance torque inputs to the Attitude Control Subsystem. The forces of any consequence to the spacecraft will be: aerodynamic, solar radiation, magnetic and gravity-gradient. The most significant torque experienced by the spacecraft at the 250 nautical mile altitude will be the aerodynamic torque. Consequently, the results of a detailed aerodynamic analysis are presented. In estimating the remaining external torques, a worst case torque due to each effect is presented.

Sufficient residual atmosphere is present at low orbital altitudes to impose significant aerodynamic torques on a moving spacecraft. However, the effects encountered at orbital altitudes differ markedly from those normally encountered by aircraft. First of all, the flow is radically different, because the low density causes the air to act more like a collection of individual particles than a continuous medium. In addition, there is no appreciable atmosphere above the orbit for distribution of the heat generated by the sun. This causes the solar atmospheric heating to vary radically with position relative to the sun which in turn creates sizable variation in the density of the local atmosphere.

Flows which are closely approximated by a collection of noninteracting particles are called Newtonian flows. They can vary from specular to diffuse

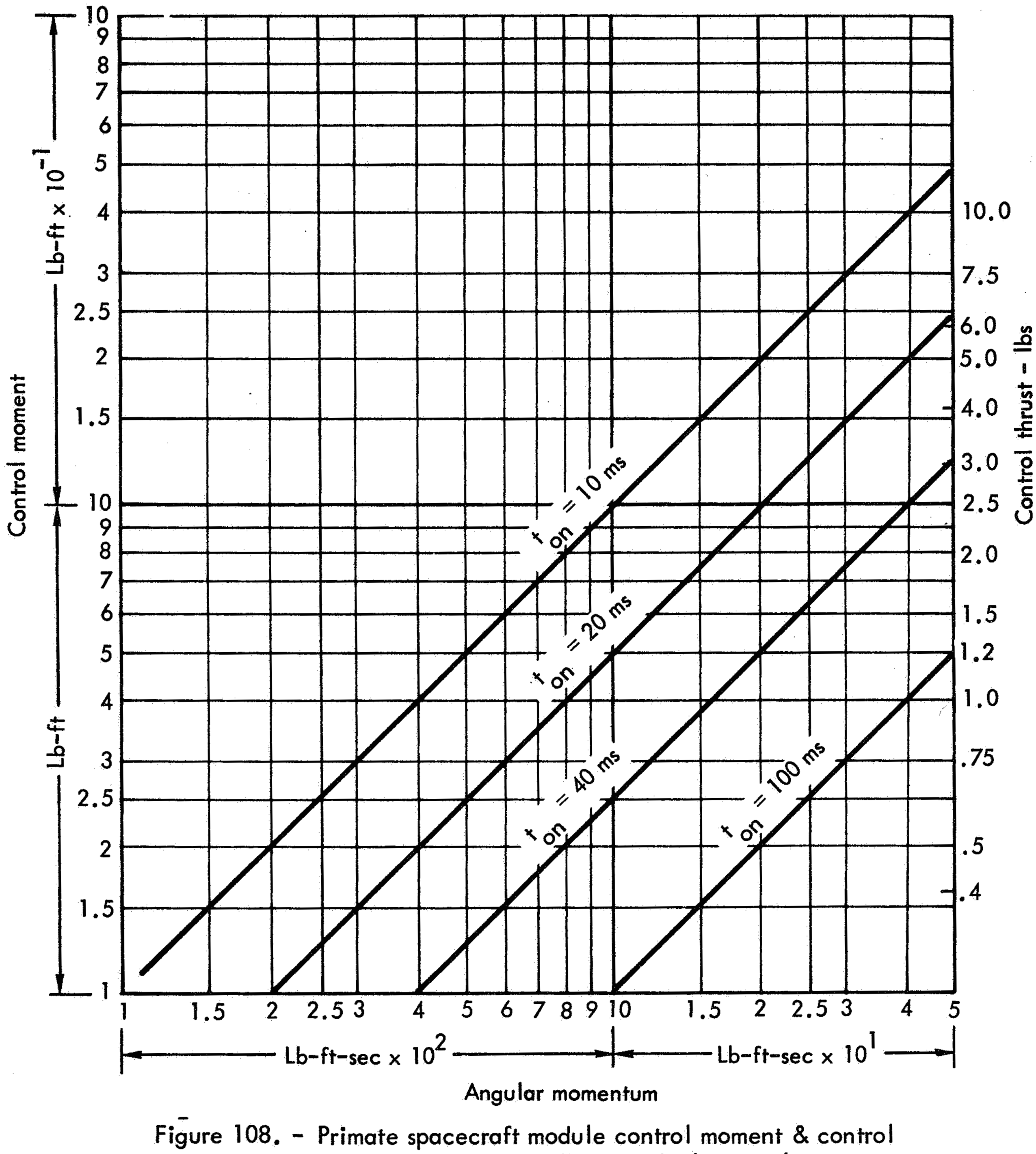


Figure 108. - Primate spacecraft module control moment & control thrust required in the limit cycle (per axis)

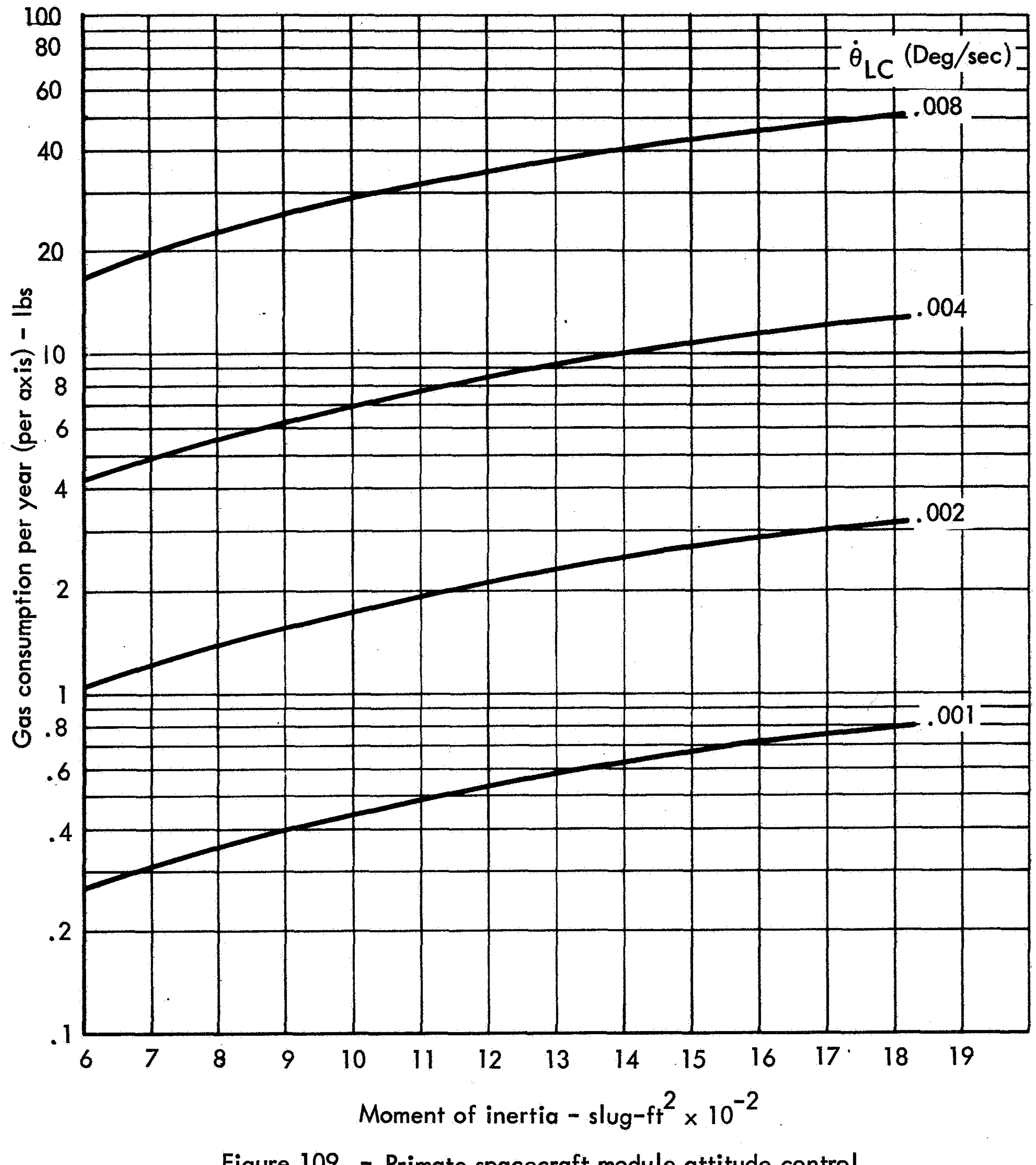


Figure 109. - Primate spacecraft module attitude control propellant (N_2) requirements

flows. The fully diffuse flow approximation appears to be the most representative of orbital conditions. Diffuse flow lift and drag coefficients for the spacecraft have been evaluated and are presented in figure 110.

The magnitude of the aerodynamic force is proportional to the dynamic pressure, which is related to the orbital velocity and the local air density. The average value of dynamic pressure throughout the 250 n mile orbit has been calculated to be 2×10^{-6} lbs/ft^2.

The magnitude of the aerodynamic force is also proportional to the wetted profile area of the spacecraft which in turn is related to the angle of attack, and sideslip. Figure 111 presents the spacecraft's wetted profile area as a function of angle of attack or sideslip.

Once the aerodynamic force is known, the resulting torque can be obtained if one can evaluate that point on the spacecraft through which the force vector tends to act. This point is commonly referred to as the center of pressure. The center of pressure location, which is also a function of angle of attack, was evaluated and is depicted in figure 112 which, for convenience, also shows the average center of gravity position. The aerodynamic force-lever arm is the distance between the center of pressure and center of gravity. Resolving this distance along the appropriate lift and drag directions produces the effective aerodynamic force-lever arms as a function of angle of attack, which is shown in figure 113. The resulting aerodynamic torque is merely the product of the aerodynamic force and the effective aerodynamic force-lever arm, and this parameter is plotted in figure 114 as a function of angle of attack. For an inertially slaved vehicle, these torques will appear to vary sinusoidally at the orbital frequency.

The incident solar radiation exerts a force on the spacecraft which is, in many respects, analagous to the aerodynamic force. The dynamic pressure associated with aerodynamic effects must be replaced by the corresponding but much smaller solar pressure. The solar lift and drag coefficients vary slightly from the corresponding aerodynamic coefficients already computed for Newtonian, free-molecular, flows. The solar force-lever arms are significantly smaller than the aerodynamic force-lever arms because of the slaved sun orientation and vehicle symmetry. The resulting maximum solar torque, 2.6×10^{-5} pound-feet is sufficiently small to the extent that it may be considered negligible.

Force fields of the earth are capable of generating two significant torques. The first to be considered is the gravity-gradient torque. The magnitude of this torque depends on three factors. It is inversely proportional to the cube of the geocentric altitude; it varies with the vehicle's size and geometry; and it varies with vehicle orientation. The maximum gravity-gradient torque experienced by the spacecraft varies throughout the mission because of the variation in the size of the moments of inertia and their relationship. This effect is shown in figure 115. It can be shown that the gravity-gradient torque on an inertially slaved vehicle lying in the orbit plane will average to zero over a complete orbit.

The second torque to be considered which is produced by an earth force field is magnetic torque. Any residual magnetism within the spacecraft will

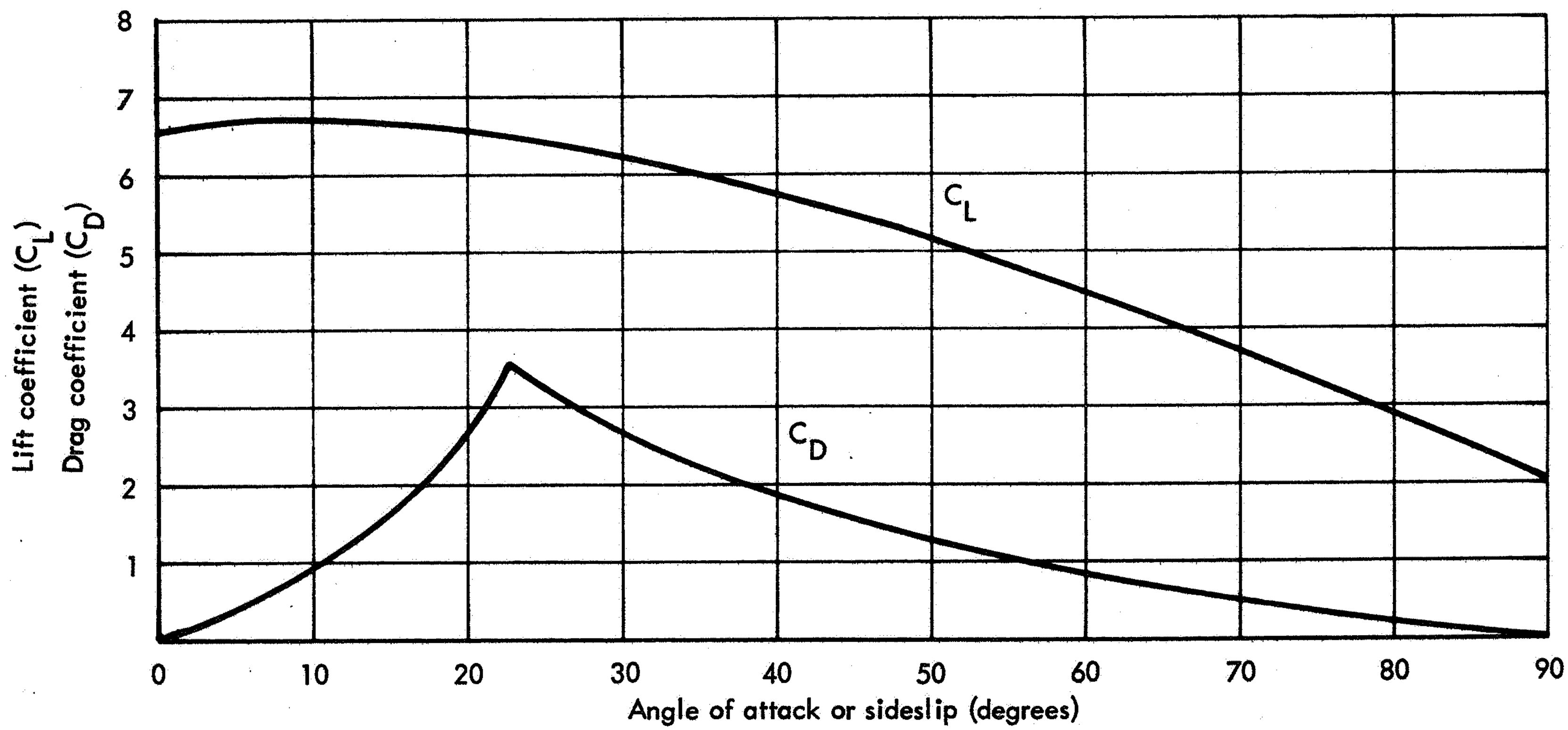


Figure 110. - Primate spacecraft module free molecular flow lift and drag coeffecients vs angle of attack or sideslip

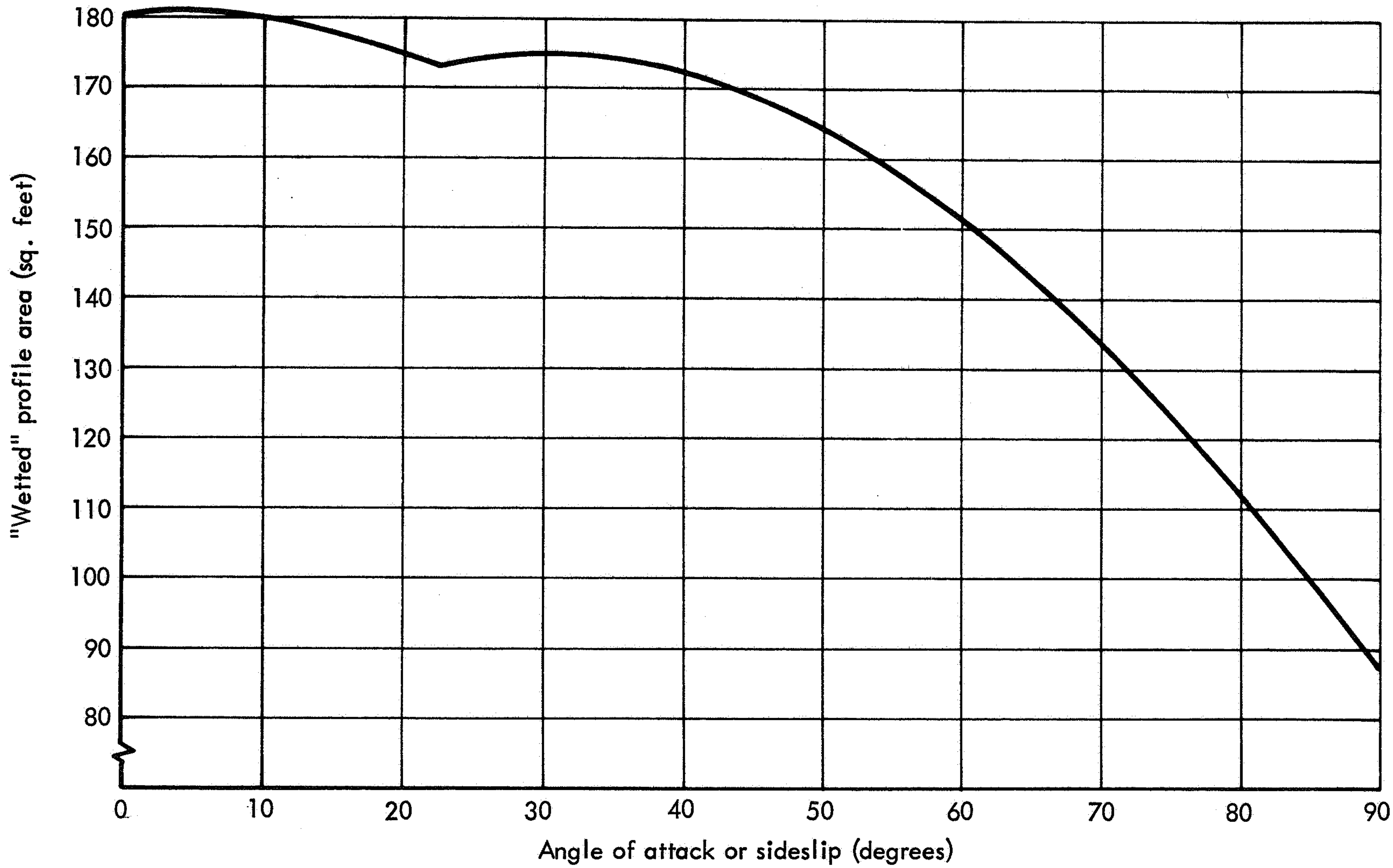


Figure 111. - Primate spacecraft module "wetted" profile area vs angle of attack or sideslip

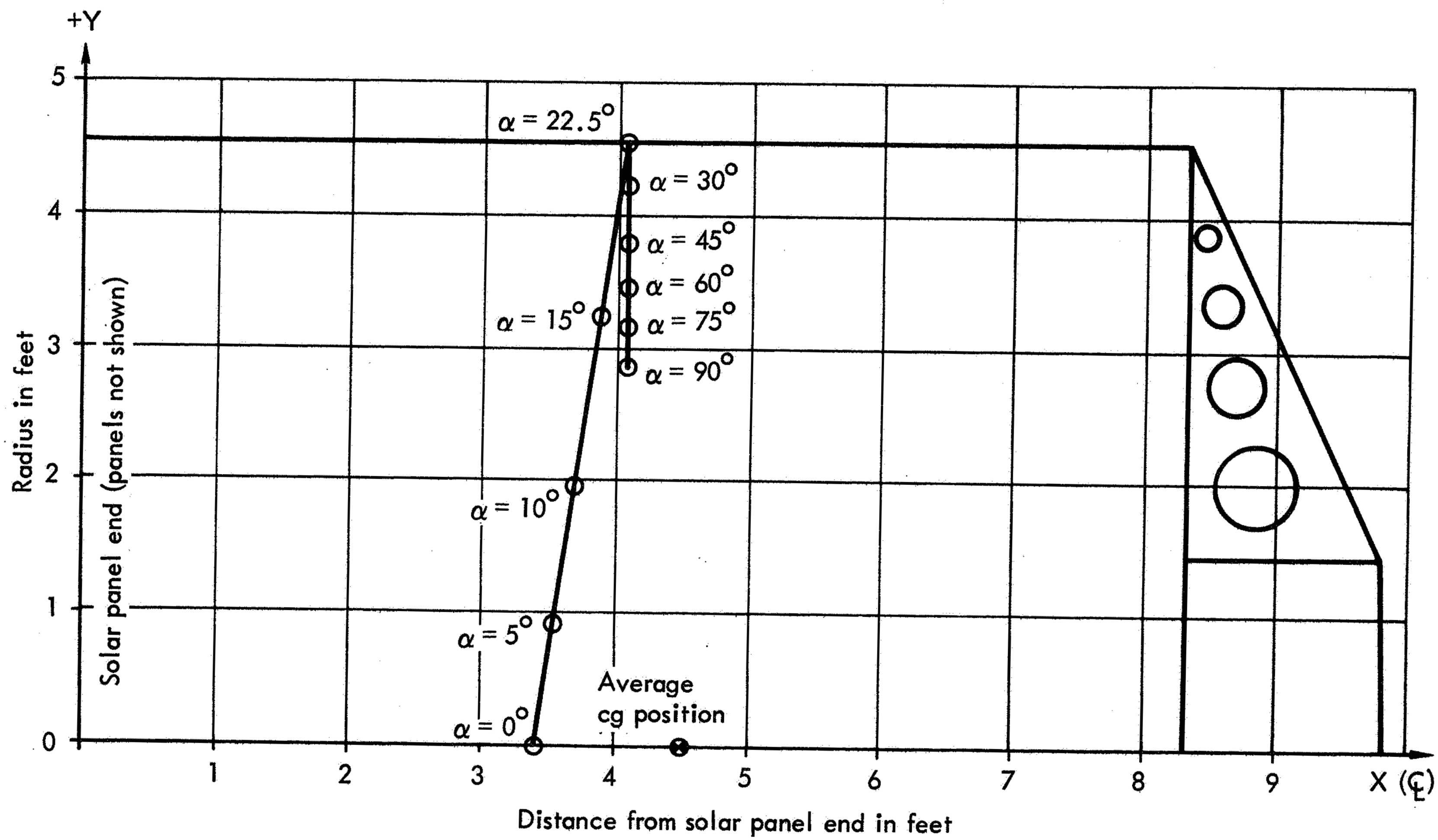


Figure 112. - Primate spacecraft module center of pressure (⊙) vs angle of attack (α) conjugate set in - yx plane (for neg. α's) due to vehicle symmetry about x-axis

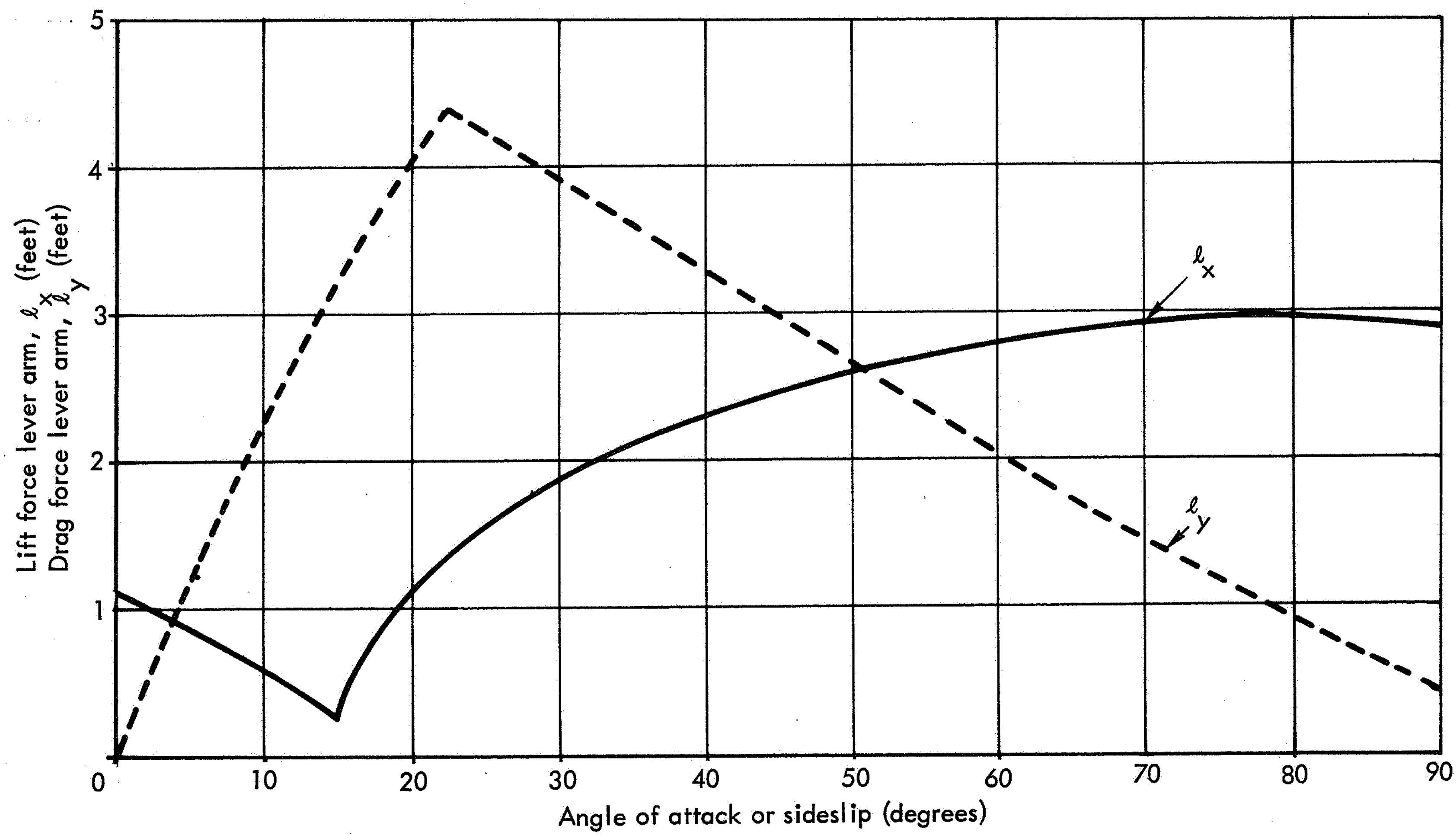


Figure 113. - Primate spacecraft module aerodynamic force lever arms vs angle of attack or sideslip

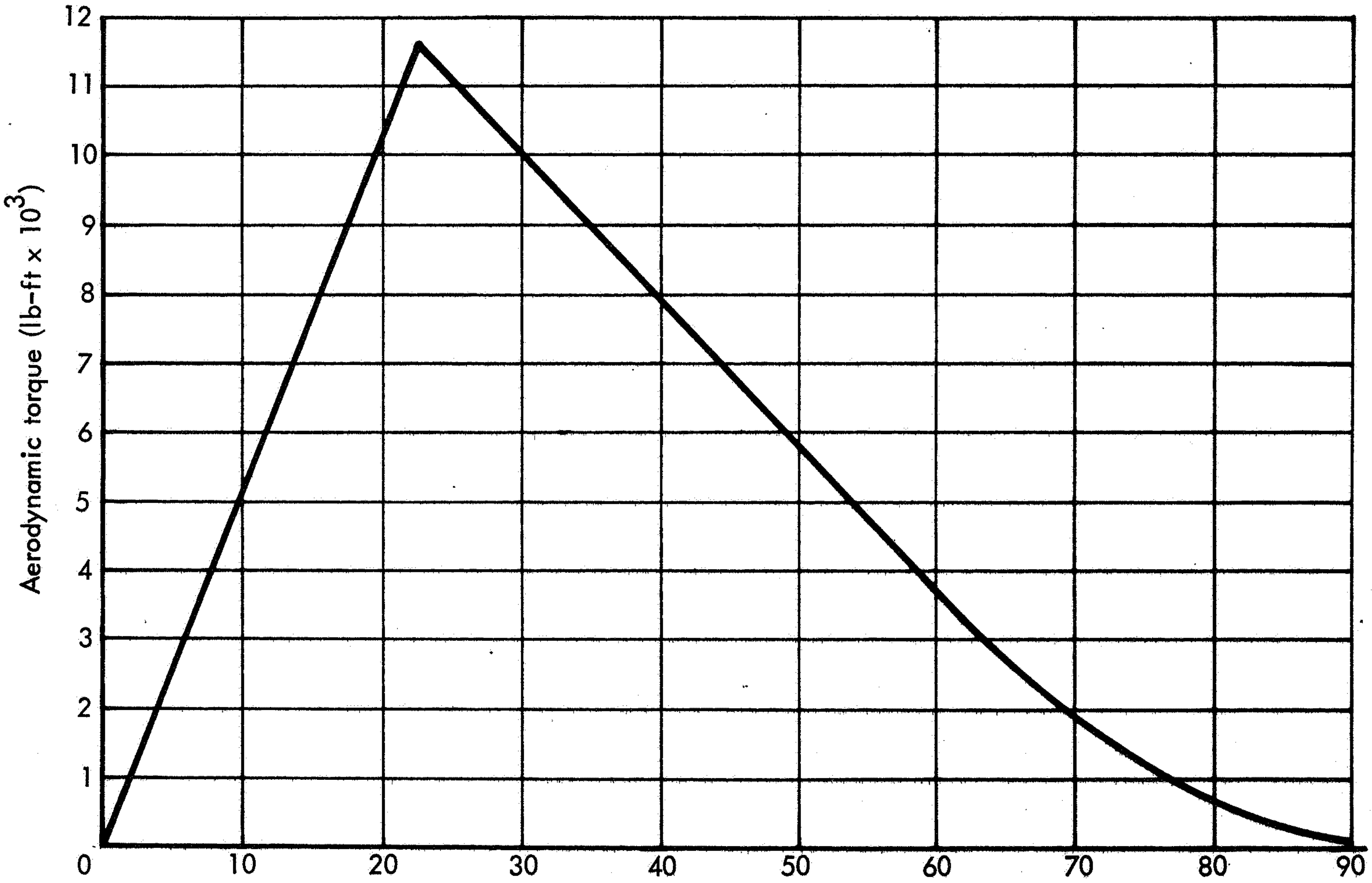


Figure 114. - Primate spacecraft module aerodynamic torque vs angle of attack or sideslip

interact with the earth's magnetic field to produce a torque which tends to align the spacecraft's magnetic axis with the earth's magnetic field. The magnitude and direction of this field vary with geocentric altitude and position of the vehicle over the earth. Therefore, the magnetic torque changes in a complicated fashion during a single orbit, as well as from orbit to orbit.

The earth's magnetic field can be approximately represented as if it were caused by a tilted dipole located at the center of the earth. This representation results in a flux density of 2.85×10^{-5} tesla. The spacecraft can also be represented by a dipole for computing the magnetic torque. The strength of the dipole and its location is best determined by measurement, but can be minimized by careful attention to design. If no magnetic control program is implemented,the "dipoles will fall where they will," and a maximum but realistic value can be assumed for purposes of computation. Certain large spacecraft have demonstrated, in orbit, values of magnetic moment as high as 10 ampere-meters^2. If the spacecraft in question were to exhibit a similar magnetic moment magnitude, the resulting torque would be approximately 2.1×10^{-4} pound-feet. This is considered to be a maximum value, and is presented as such in figure 115.

A summary of the external disturbances considered in this study are presented in figure 116.

Internal disturbances: Internal disturbances originate from such factors as rotating machinery, moving structures, and primate motion. Rotating machinery include the main circulation fan, smaller fan, pump and feeding mechanism. The maximum cumulative torque generated by rotating machinery is calculated to be equal to 2 pound-feet. This torque is not continuous and the effect from the largest contributor, that of the circulating fan, is experienced for approximately 0.5 second after start-up.

The moving wall would, by itself, exert a net torque of zero on the spacecraft each time it cycles. However, it is likely that the wall will carry the primate toward the top of the cage. In addition, the wall's motion may be somewhat impaired due to a collection of debris on the surface of the cage floor. The net torque required for the moving wall under these conditions is estimated to be 0.2 pound-feet acting for a duration of 0.5 second.

The worst contemplated effect that the primate's motion could have on the spacecraft's motion would result from a unidirectional, circulating walk. Assuming a rate of travel of 2 mph, and an acceleration time of 0.5 second, the resulting torque would be approximately 2.5 pound-feet for a duration of 0.5 second.

Of the above mentioned effects, the continuously rotating machinery would produce a continuous gyroscopic torque (in addition to the instantaneous torque) on the spacecraft. The magnitude of this torque is proportional to the product of the angular momentum of the machinery and the angular inertial velocity of the spacecraft. This gyroscopic torque is calculated to be a maximum of 10^{-4} pound-feet.

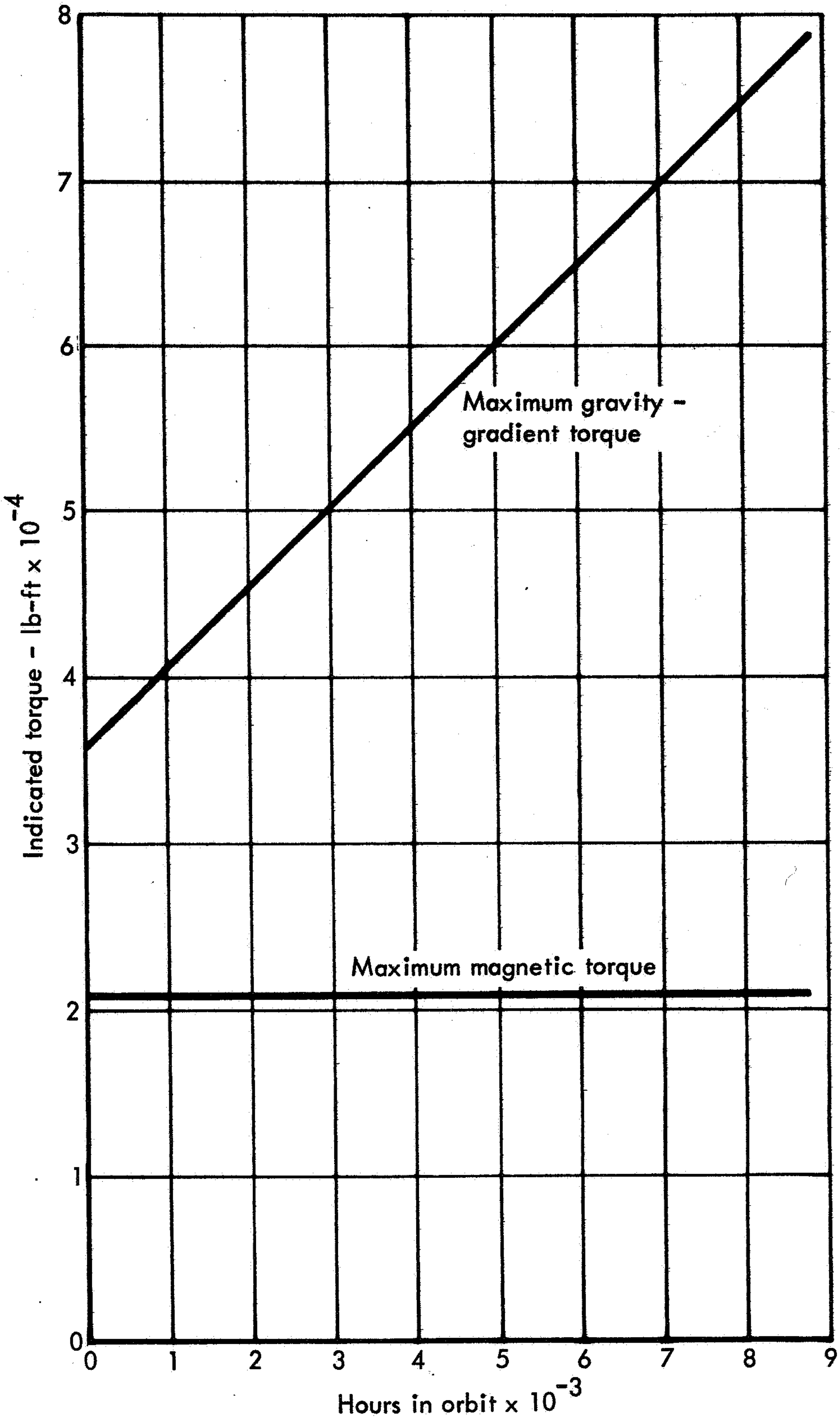


Figure 115. - Primate spacecraft module disturbance torques generated by earth's force fields

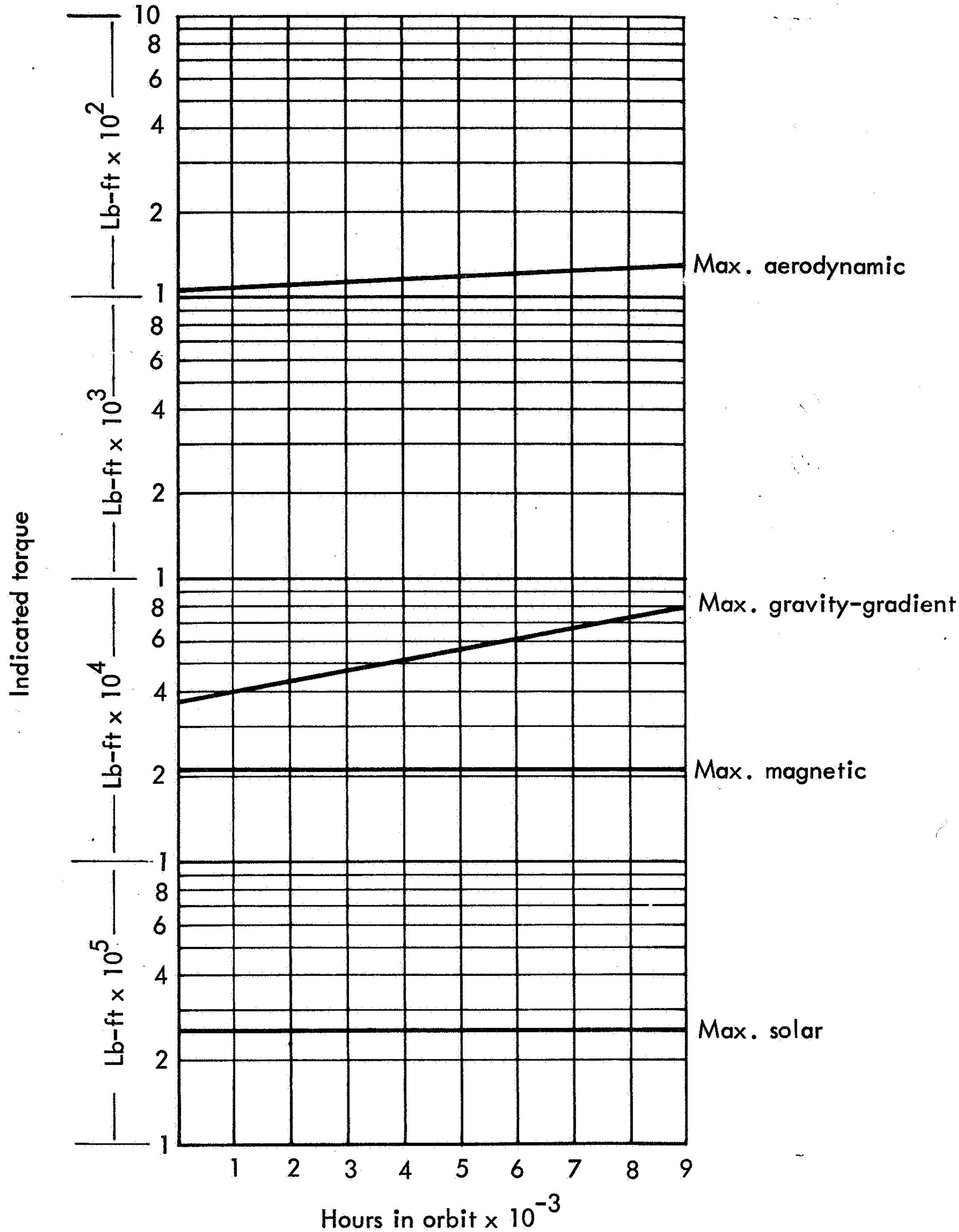


Figure 116. - Primate spacecraft module maximum external distrubances

Pitch or yaw maneuver: The Attitude Control Subsystem has a maneuvering capability, which may be utilized to facilitate docking, or to reacquire the sun, if ever necessary. The two important factors concerning any designated maneuver are the time it takes to perform the maneuver and the propellant expended in doing so; one factor being a function of the other. If there does not exist a stringent time factor for performing the maneuver, any resonable maneuver can be accomplished with negligible propellant expendature. For example, a 180 degree pitch or yaw maneuver can be accomplished by using two single minimum pulses. The time to perform the manuever is approximately 16.5 hours and the gas consumption of 2×10^{-3} pounds.

Angular rate limit: The constraint that the primates shall not experience a continuous acceleration in excess of 0.001 g places an upper limit on the angular rate of the spacecraft. The maximum allowable rate corresponding to this constraint is 5.6 degrees/second about any axis. The Attitude Control Subsystem precludes such an occurrence during normal operation by nature of its implementation.

Advance development areas: No special advanced development is anticipated beyond the present state of the art for the selected mechanization.

Preliminary equipment list: A preliminary equipment list is presented in table 56. Every item is readily available and has demonstrated operational capability in space and defense programs.

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NORTHROP SYSTEMS LABORATORIES

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Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

December 1, 1967

TABLE 56. - ATTITUDE CONTROL SUBSYSTEM PRELIMINARY EQUIPMENT LIST (8000)

Item No.	Description	Suggested Manufacturer	Part No.	Quantity per Spacecraft
1	TARS package	Nortronics; Honeywell or Kearfott		2
2	Sun sensors	TRW or Adcole		8
3	Gyro signal amps	Nortronics		3
4	Sun sensor amps	Nortronics		2
5	Signal conditioners	Nortronics		3
6	Valve driver amps	Nortronics		6
7	Schmitt Triggers	Nortronics		3
8	Reaction jets (Nozzles)	Kidde		12
9	Solenoid valves	Kidde		12
10	Charging valves	Kidde		2
11	Relief valves	Kidde		2
12	Squib valves	Kidde		2
13	Filters	Kidde		2
14	Regulators	Kidde		2
15	Pressure reducers	Kidde		2
16	Storage tanks with nitrogen gas	Kidde		2

APPENDIX A

PRIMATE SPACECRAFT BASIC DIMENSIONS

The basic dimensions and definitions shown on the following pages shall be used for the Baseline Spacecraft design.

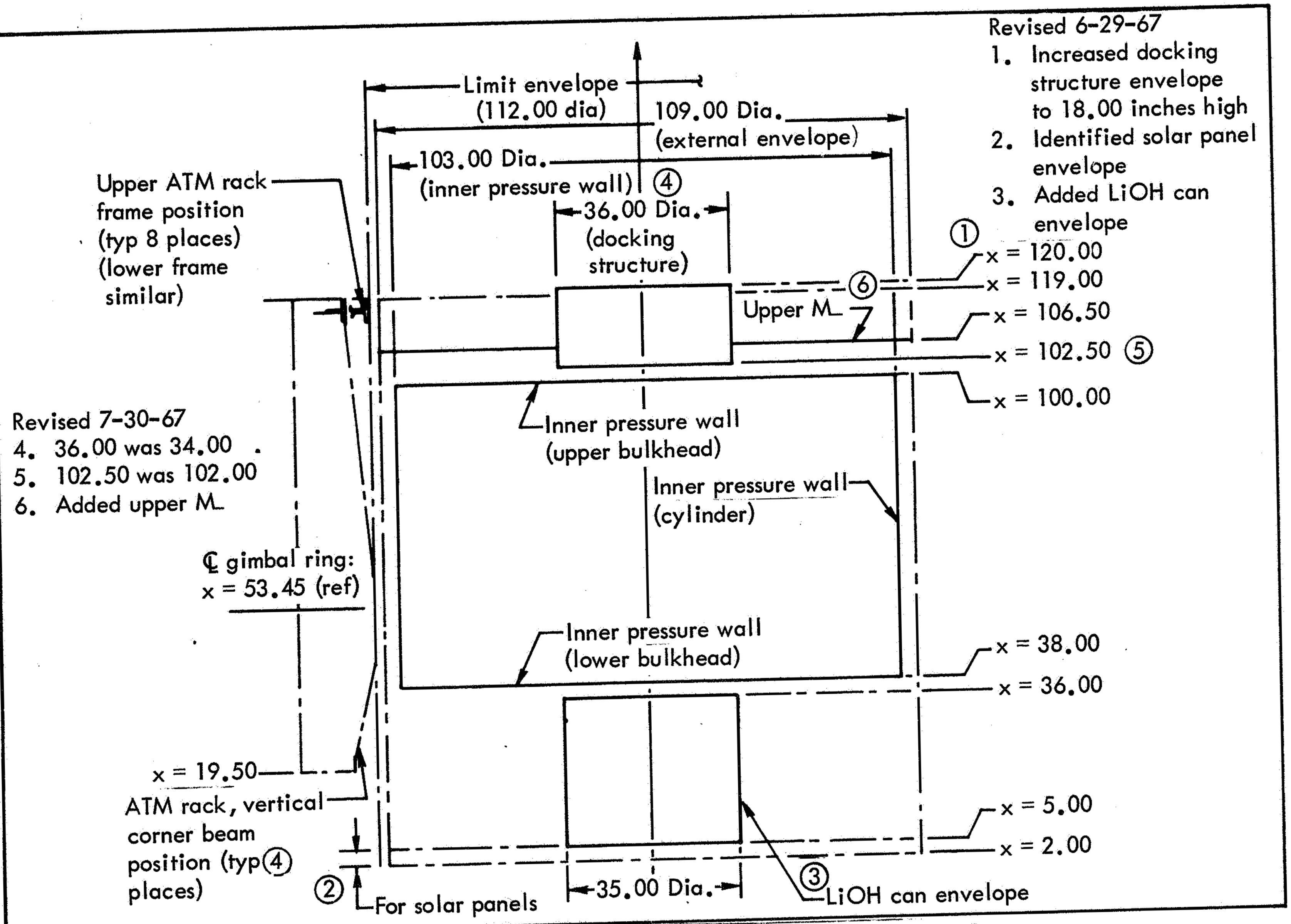


Figure A1. - Sideview basic dimensions of primate spacecraft

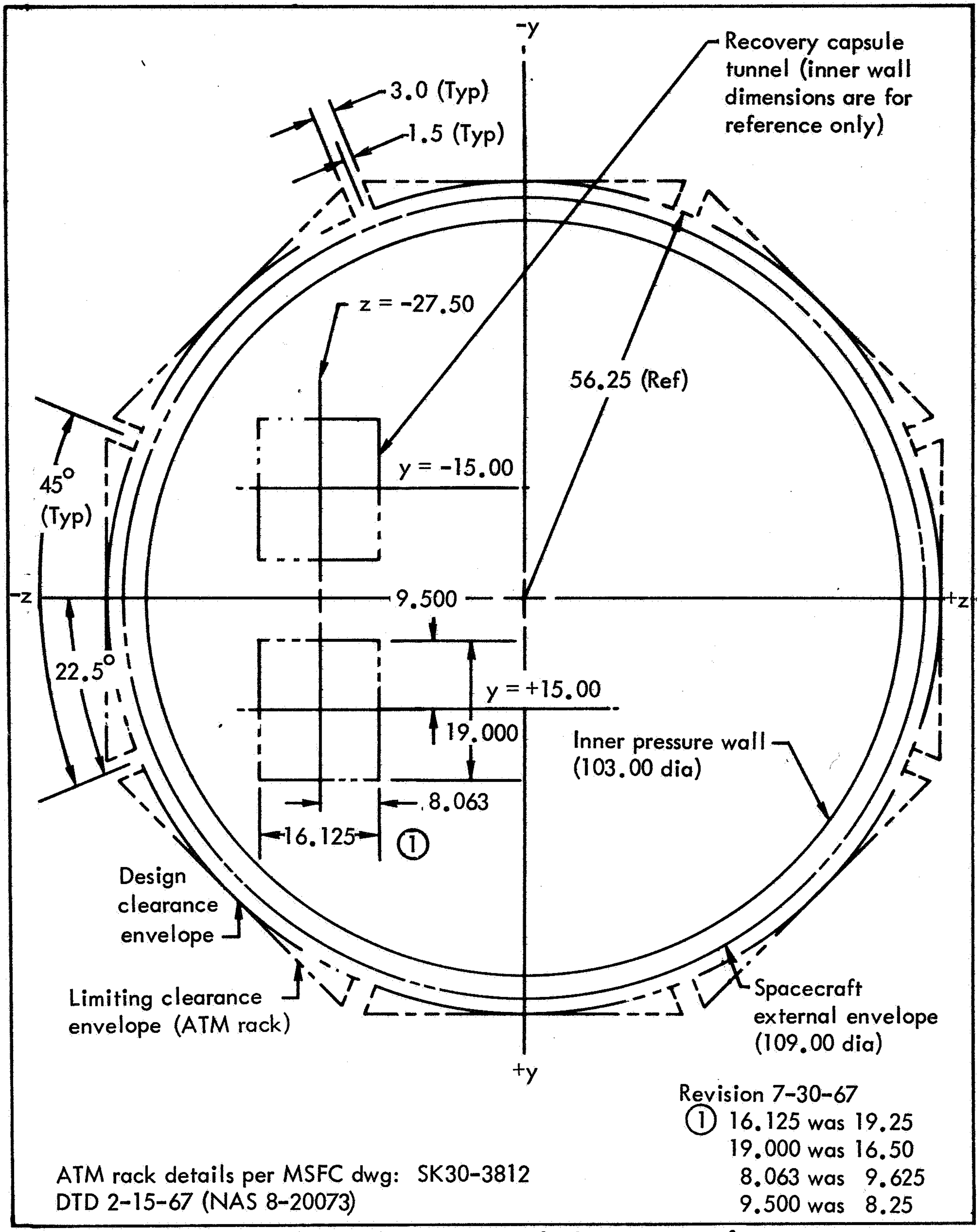


Figure A2. - Basic dimensions of primate spacecraft

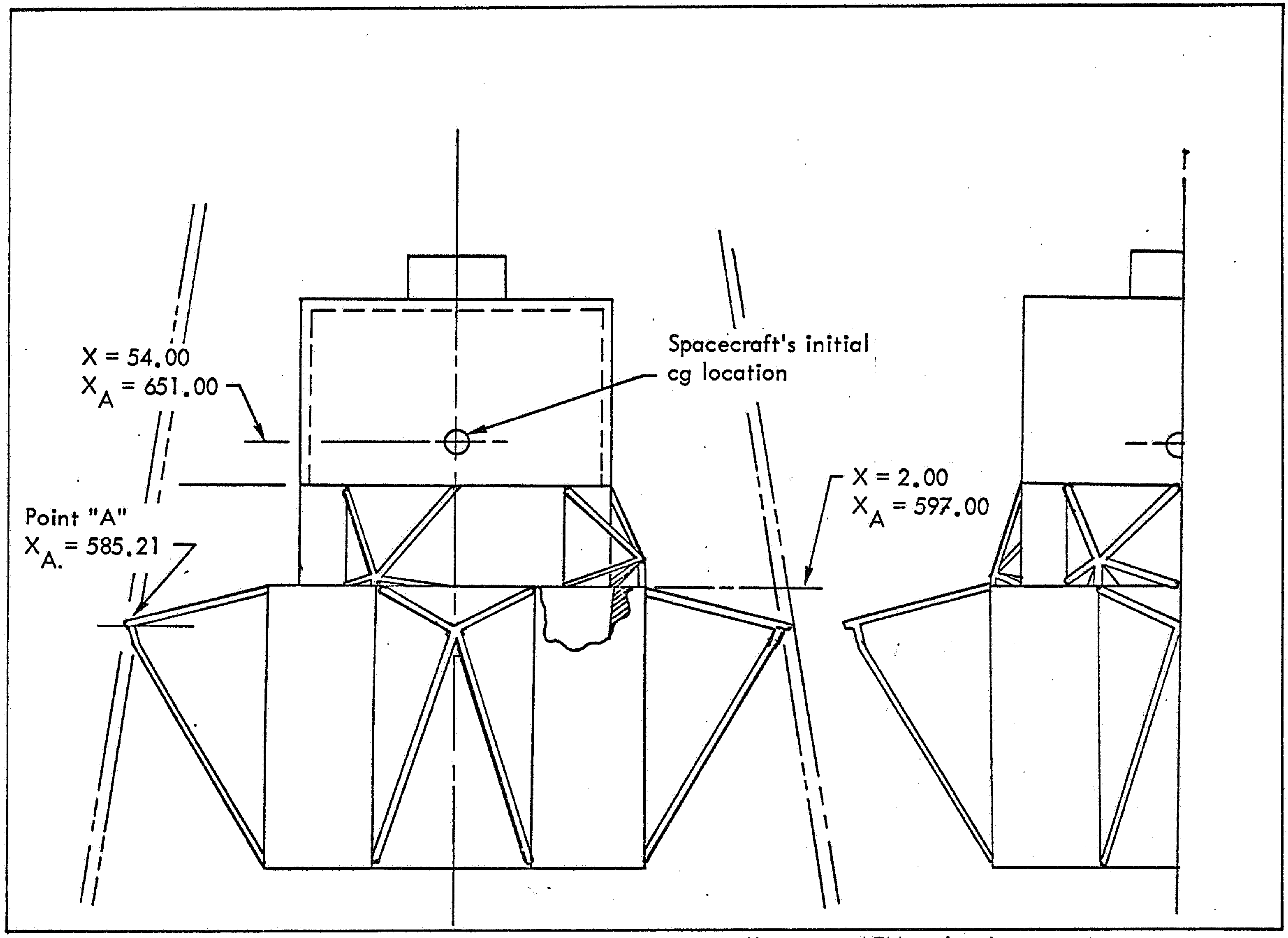


Figure A3. - Primate spacecraft design point installation on ATM rack using LEM ascent stage attach points (ref MSFC dwg SK30-3812 DTD 2-5-67).

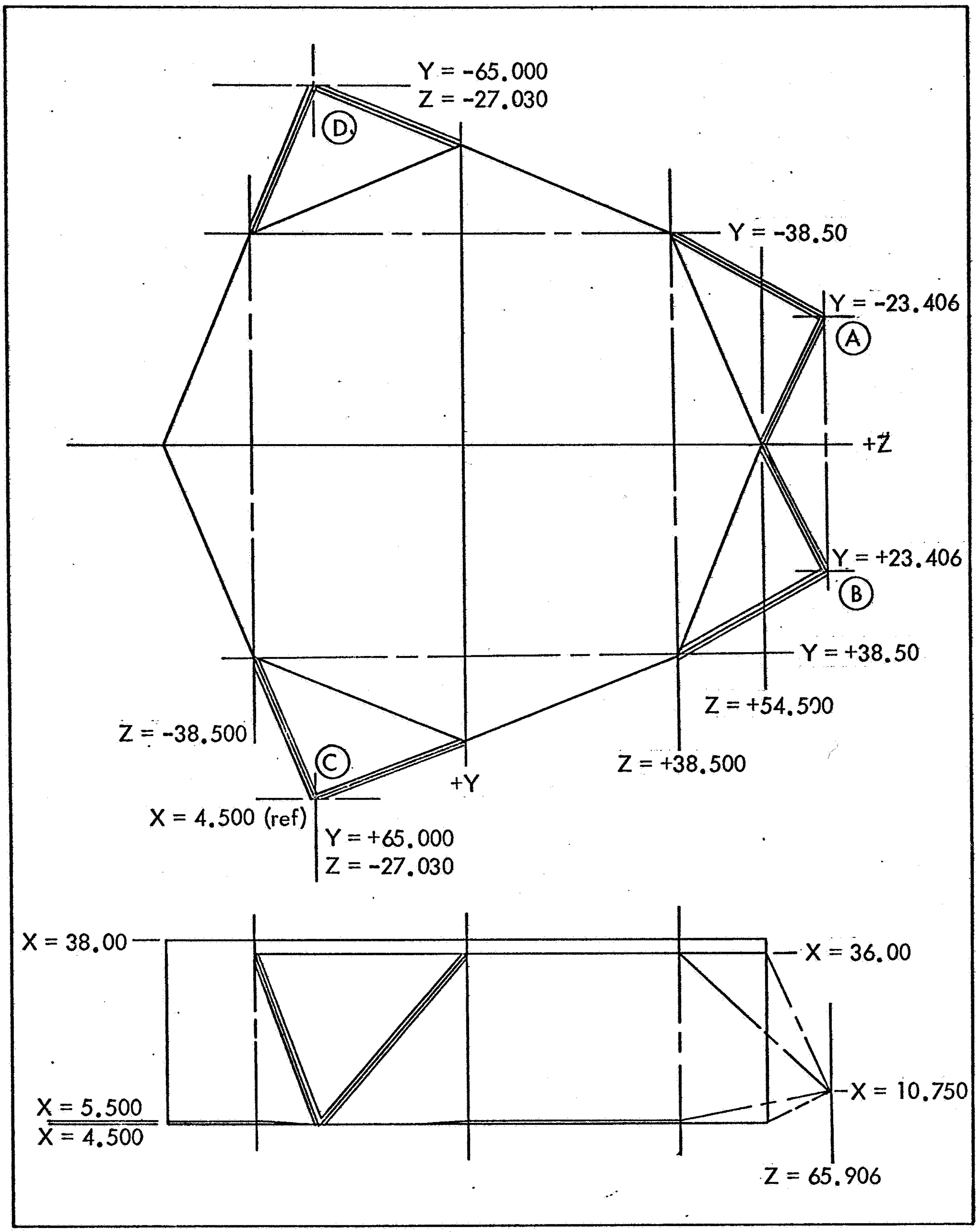


Figure A4. - Primate S/C to ATM attach point geometry

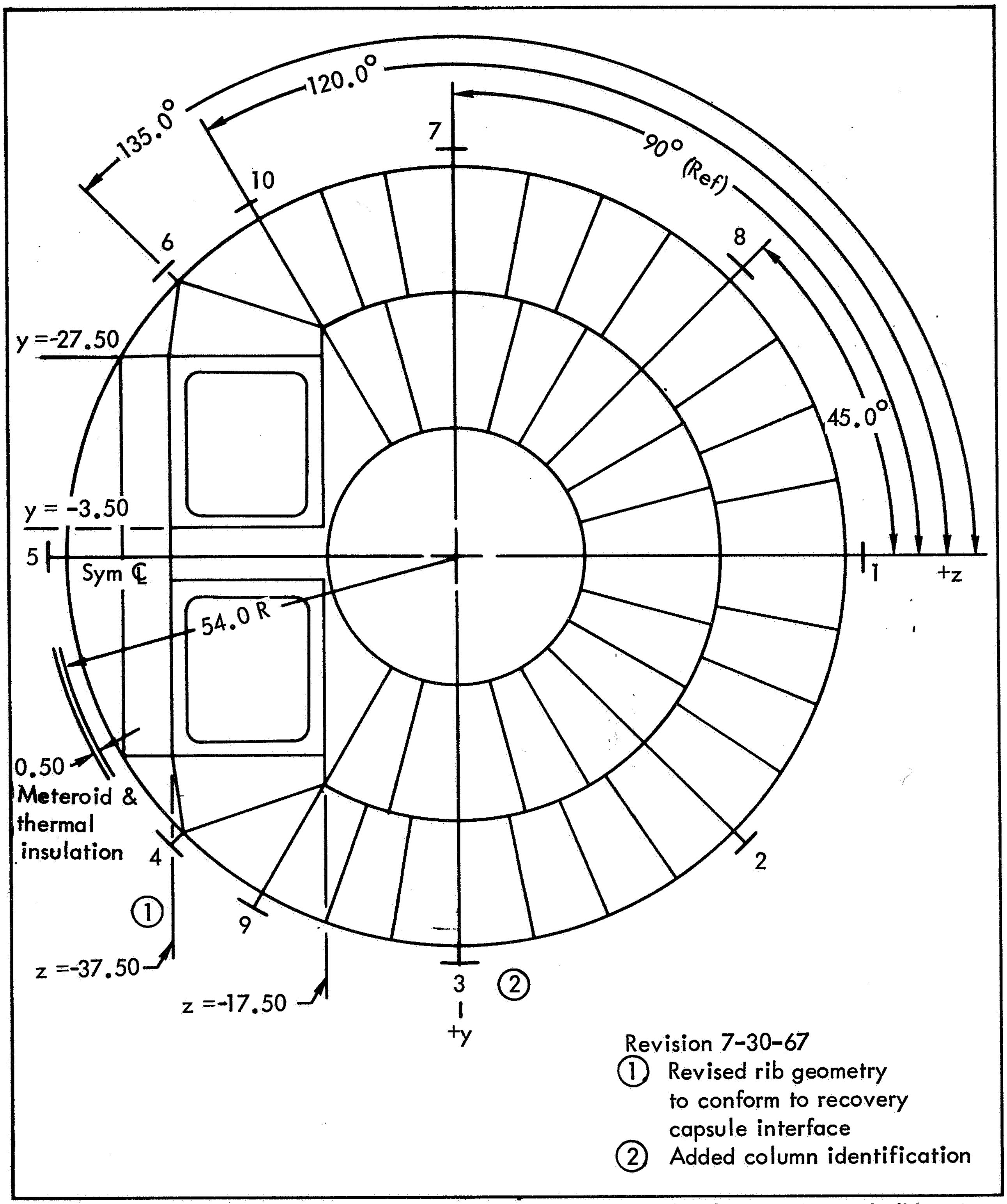


Figure A5. - Geometry: upper bulkhead beams and vertical columns. Lower bulkhead has uniform pattern similar to $+y_1$ - +z quadrant primate spacecraft

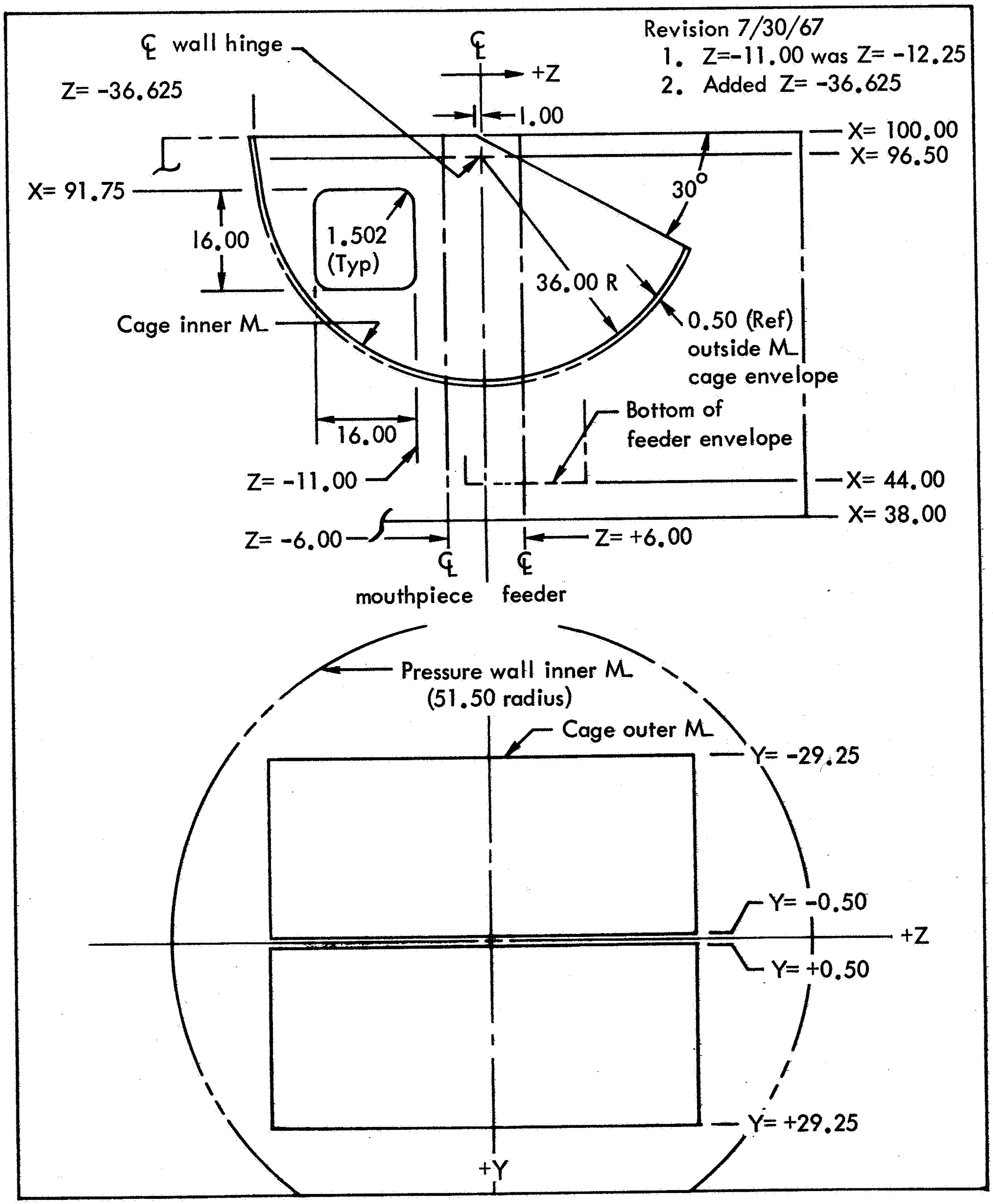


Figure A6. - Primate spacecraft cage geometry

APPENDIX B

REACTION EQUATIONS

Reaction Equations for the Baseline Spacecraft attach points are given in table B 1. The following definitions apply:

R = Reaction

A = Attach point in -y, +z quadrant

B = Attach point in +y, +z quadrant

C = Attach point in +y, -z quadrant

D = Attach point in -y, -z quadrant

x = Launch axis; vertical at lift-off (roll axis)

y = Lateral axis, (pitch)

z = Lateral axis, (yaw)

η = Load factor in g's

W = Spacecraft weight (= 5000 pounds for baseline loads)

θ = Displacement of lateral load factor in the y - z plane and measured clockwise from the +z axis

ℓ = Subscript, lateral

Lateral load factor is resolved to y and z coordinates as follows:

$\eta_y = \ell \text{ Sin } \theta$

$\eta_z = \ell \text{ Cos } \theta$

TABLE B1. - DESIGN POINT SPACECRAFT ATTACH POINT REACTIONS

Attach point	Vertical load (x axis)	Lateral load (y axis)	Lateral load (z axis)
A	$R_{A_x} = (0.1455\,\eta_x + 0.381\,\eta_y - 0.266\,\eta_z)\,W$	$R_{A_y} = +0.1455\,\eta_y W$	$R_{A_z} = +0.25\,\eta_z W$
B	$R_{B_x} = (0.1455\,\eta_x - 0.381\,\eta_y - 0.266\,\eta_z)\,W$	$R_{B_y} = +0.1455\,\eta_y W$	$R_{B_z} = +0.25\,\eta_z W$
C	$R_{C_x} = (0.3545\eta_x - 0.2435\eta_y + 0.266\,\eta_z)W$	$R_{C_y} = +0.3545\,\eta_y W$	$R_{C_z} = +0.25\,\eta_z W$
D	$R_{D_x} = (0.3545\eta_x + 0.2435\eta_y + 0.266\,\eta_z)W$	$R_{D_y} = +0.3545\,\eta_y W$	$RD_z = +0.25\,\eta_z W$

APENDIX C

ORBIT TRANSFER LOADS

Loads are induced into the Primate Spacecraft through the docking collar by the Apollo Command Service Module during Spacecraft Propulsion System firing for transfer from the parking to the operational orbit. The Spacecraft Propulsion System engine provides 21,900 pounds of thrust and has a maximum gimbal angle of about 8.5°. The following analysis is based on steady-state values of this thrust to provide limit loads for the preliminary stress analysis. Vehicle system elements are described in table C-1A (Properties of Vehicle System Elements) and table C-1B (Analytical Model Geometry). The Primate Spacecraft weight was assumed to range from 4000 to 8000 pounds, with a baseline design point for structural loads being taken as 5000 pounds. Moments, and Axial and Lateral loads were derived by examining the following four cases:

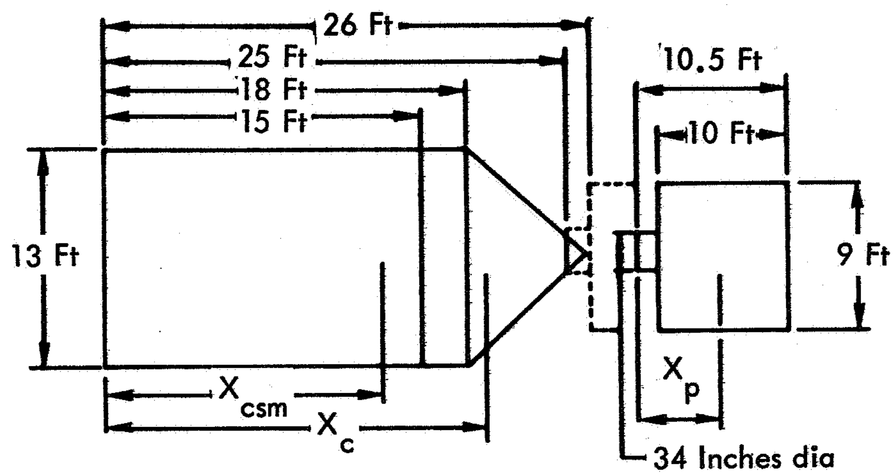
(1) CSM initial weight, Gimbal angle = 0°
(2) CSM final weight, Gimbal angle = 0°
(3) CSM initial weight, Gimbal angle = 8.5°
(4) CSM final weight, Gimbal angle = 8.5°

TABLE C1. - PROPERTIES OF VEHICLE SYSTEM ELEMENTS

Element	Weight (Lb)	$I_z = I_y$ (slug-ft^2)	Remarks
CSM	21500	---	Dry
CSM	23100	34000*	With 1600 lb of residuals
CSM	25117	38000*	With 1600 lb of residuals plus 2017 lb of orbit transfer fuel
SIVB + IU	29030	---	With residuals
ATM Rack	2200	---	---
SLA	3990	---	---

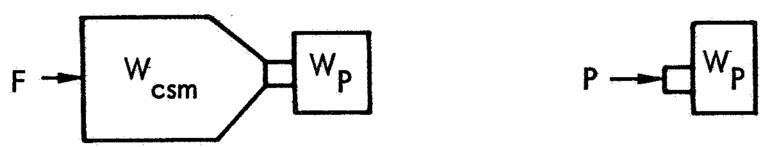
* Approximations based on homogeneous body increments.

TABLE C2. - ANALYTICAL MODEL GEOMETRY



Center of Gravity	Location	
	With 1600 lb residuals	With 1600 lb residuals plus 2017 lb of maneuvering fuel
X_{csm}	12.78	12.1
X_p	5.5	5.5
X_c	15.92	15.15

<u>Case 1.</u> CSM initial weight, Gimbal angle = 0°



Acceleration of the spacecraft complex, a_c, is:

$$a_c = F/m = Fg/(W_{csm} + W_p) \qquad (C1)$$

Acceleration of the Primate Spacecraft is:

$$a_p = F/m = Pg/W_p$$

Combining (1) and (2): (C2)

$$\frac{Pg}{W_p} = \frac{Fg}{(W_{csm} + W_p)}$$

and solving for P:

$$P = \frac{F\ W_p}{(W_{csm} + W_p)} \qquad \text{(C3)}$$

Substituting appropriate values in (3)

$$P = \frac{(21900)\ (5000)}{(25117 + 5000)} = 3630 \text{ lb} \qquad \text{(C4)}$$

Acceleration of the spacecraft complex [from (C1)]:

$$a_c = \frac{21900}{30117} = 0.729 \text{ g}$$

Case 2. CSM final weight, Gimbal angle = 0°

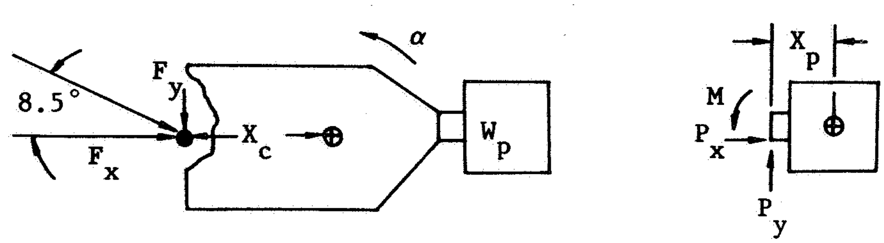
From (C3), Case 1:

$$P = \frac{(21900)\ (5000)}{(23100 + 5000)} = 3900 \text{ lb}$$

Acceleration of the spacecraft [from (C1)] :

$$a_c = \frac{21900}{28100} = 0.78 \text{ g}$$

Case 3. CSM initial weight, Gimbal angle = 8.5° (Hard over engine)



$$F_x = 21900 \cos 8.5° = 21700 \text{ lb} \qquad \text{(C5)}$$

$$F_y = 21900 \sin 8.5° = 3240 \text{ lb} \qquad \text{(C6)}$$

The loads are computed as follows:

(a) Find net linear acceleration at "O" = a_n

(b) Then $P_y = \frac{W}{g} a_n$, $P_x = \frac{F_x g}{(W_{csm} + W_p)}$, $M = \left(\frac{W_p}{g} a_n x_p\right) + \left(I_{cgp} \alpha\right)$

$$a_n = a + a_y \tag{C7}$$

$$a = \frac{F_y g}{W_{csm} + W_p} = \frac{(3240)\ (32.2)}{W_{csm} + W_p} = \frac{104200}{W_{csm} + W_p} \tag{C8}$$

$$a_y = r\,\alpha = (30.5 - x_c)\ \frac{(3240)\ \ (X_c)}{I_{cg}} \tag{C9}$$

where I_{cg} is the pitch (y axis) or yaw (z axis) moment of inertia of the CSM/PSC. I_{cg_y} is assumed equal to I_{cg_z}

Substituting (C8) and (C9) in (C7) and taking P_y as the positive direction:

$$a_n = \frac{(30.5 - x_c)\ (3240)\ (w_c)}{I_{cg}} - \frac{104200}{W_{csm} + W_p} \tag{C10}$$

x_c and I_{cg} are defined as follows:

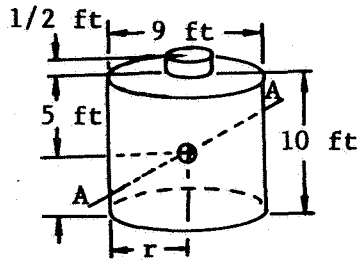
$$x_c = \frac{(W_{csm})\ (x_{csm}) + (W_p)\ (30.5)}{W_{csm} + W_p} \tag{C11}$$

$$I_{cg} = I_{csm} + \frac{W_{csm}}{g}\ (x_c - x_{csm})^2 + 0.329\ W_p + (30.5 - x_c)^2\ \frac{W_p}{g} \tag{C12}$$

The constant 0.329 in the third term is derived as follows:

1. Assumptions

a. Homogeneous right circular cylinder:



W_p = weight of Primate Spacecraft

Note: cg position is assumed fixed within the model weight range of 4000 to 8000 pounds

2. I_{cg} about A-A

$$I_{cg} = \frac{m}{12}(3r^2 + h^2) = \frac{W_p}{(32.2)(12)}(3.3^2 + 10^2) = \frac{127\ W}{(32.2)(12)}$$

$$= 0.329\ W\ \text{slug ft}^2$$

3. Check:

Per Attitude Control Trade Study:

W = 4057.2

$I_x = I_y$ = 1390 slug-ft^2 (solar panels extended)

= 1300 to 1350 slug-ft^2 (solar panels folded)

c.g. @ 66.0 inches from docking interface; 5.5 feet

From Model

@ W = 4057.2 Chk

$I_x = I_y$ = (0.329) (4057.2) = 1334 slug-ft^2 (solar panels folded) Chk

cg @ 5.5 ft from docking interface Chk

Making the appropriate substitution in (C11) and (C12):

$$x_c = \frac{(25117)(12.1) + (5000)(30.5)}{(25117 + 5000)} = \frac{304000 + 152500}{30117}$$

$$= \frac{456500}{30117} = 15.15\ \text{ft}$$

$$I_{cg} = 38000 + \frac{(25117)}{(32.2)} (15.15 - 12.1)^2 + (0.329)(5000)$$

$$+ (30.5 - 15.15)^2 \frac{5000}{32.2} = 38000 + 7256 + 1645 + 36600$$

$$= 83501 \text{ slug-ft}^2$$

Inserting these values in equation (C10):

$$a_n = \frac{(30.5 - 15.15)(3240)(15.15)}{83501} - \frac{104200}{30117} = 9.03 - 3.46$$

$$= 5.57 \text{ ft/sec}^2$$

Then $P_y = \frac{W_p}{g} a_n = \frac{5000}{32.2} (5.57) = 865$ lb

$$M = \left(\frac{5000}{32.2}\right)(5.57) (5.5) + \frac{(1645)(3240)(15.15)}{83488}$$

$$= 4760 + 967 = 5727 \text{ lb-ft}$$

or, in lb-inches;(5727)(12) = 68700 lb-inches

The component of P_x due to the moment

$$= P_{x_m} = \frac{68700}{34.0} = 2020$$

$$P_x = P_{x_{F_x}} + P_{x_m} \tag{C13}$$

From (C3) $P_{x_{F_x}} = \frac{F_x W_p}{(W_{csm} + W_p)} = \frac{(21700)(5000)}{30117} = 3600$

Then, from (C13) $P_x = 3600 + 2020 = 5620$ lb

Summary, Case 3: $P_x = 5620$ lb

$P_y = 865$ lb

$M = 68700$ lb-in

Case 4. CSM final weight, Gimbal angle 8.5° (Hard over engine)

$F_x = 21700$ lb [from (C5)]

$F_y = 3240$ lb [from (C6)]

$x_c = 15.92$ [from table (C2)]

$x_{csm} = 12.78$ [from table (C2)]

From (C12):

$$I_{cg} = 3400 + \frac{23100}{32.2}(15.92 - 12.78)^2 \times (0.329)(5000)$$

$$+ (30.5 - 15.92)^2 \frac{5000}{32.2} = 34000 + 7070 + 1645 + 33000$$

$$= 75715 \text{ slug-ft}^2$$

From (C10):

$$a_n = \frac{(30.5 - 15.92)(3240)(15.92)}{75715} - \frac{104200}{28100} = 9.93 - 3.71$$

$$= 6.22 \text{ ft/sec}^2$$

Then: $P_y = \left(\frac{5000}{32.2}\right) 6.22 = 966$

$$M = \frac{(5000)(6.22)(5.5)}{32.2} + \frac{(1645)(3240)(15.92)}{75715} = 5310 + 1121$$

$$= 6431 \text{ lb-ft}$$

or, in lb-in; $(6431)(12) =$ [illegible] 200 lb-in

From (C3)

$$P_{x_{F_x}} = \frac{(21700)(5000)}{(28117)} = 3860 \text{ lb}$$

$$P_{x_m} = \frac{77200}{34} = 2270 \text{ lb}$$

$$P_x = P_{x_{F_x}} + P_{x_m} = 6130 \text{ lb}$$

Summary, Case 4:

$P_x = 6130$ lb

$P_y = 966$ lb

$M = 77200$ lb-in

Discussion

As one would intuitively expect, the maximum loads during orbit transfer on the docking collar of the Primate Spacecraft occur with a hard-over engine at the end of the Spacecraft Propulsion System burn. These loads are compared in table C3 with the loads imposed during docking and those experienced by the Lunar Module.

TABLE C3. - COMPARISON OF PRIMATE SPACECRAFT DOCKING LOADS

Load	Limit load Magnitude & event		Lunar module
	Docking	Orbit transfer	
P_1(lb)	~2200	6130	Loads increase with increasing weight of docked payload: $\frac{\text{LEM}}{\text{Primate S/C}} \frac{25000}{5000} = 5$
P_2(lb)	~2600	966	See figures C1 and C2 for trends and NAA/GAEC Interface Control Document ICD MH01-05050-414, 4/4/66 for detailed loads summary
M(lb-in)	~35000	77200	

See figure 32, Requirements Section of Structure and Mechanical System Section for pre-separation pressure loads

The steady-state accelerations imposed on the spacecraft are low (less than 0.8g), and are not critical to the design.

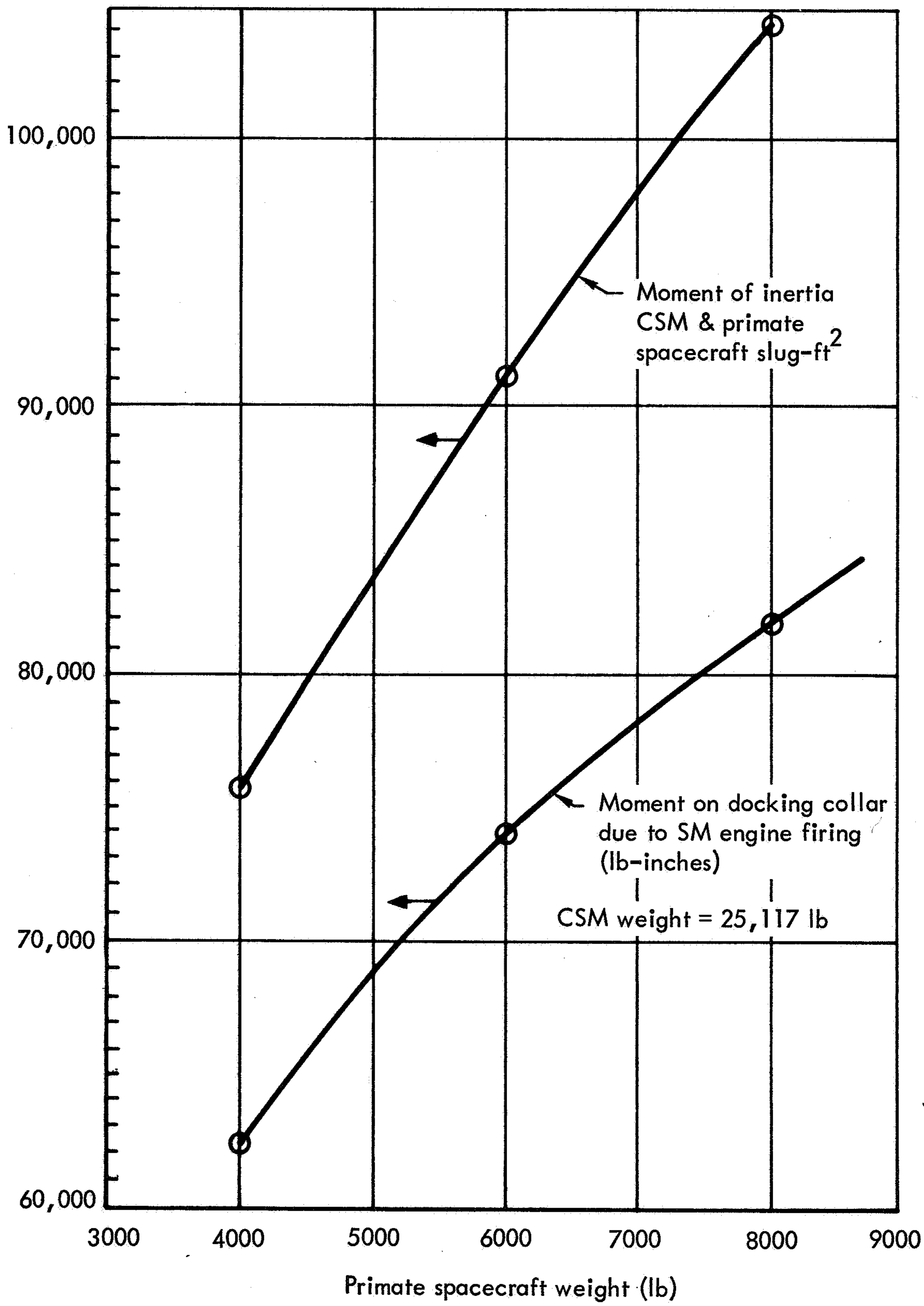


Figure C1. - Moment of inertia and docking collar moment trends

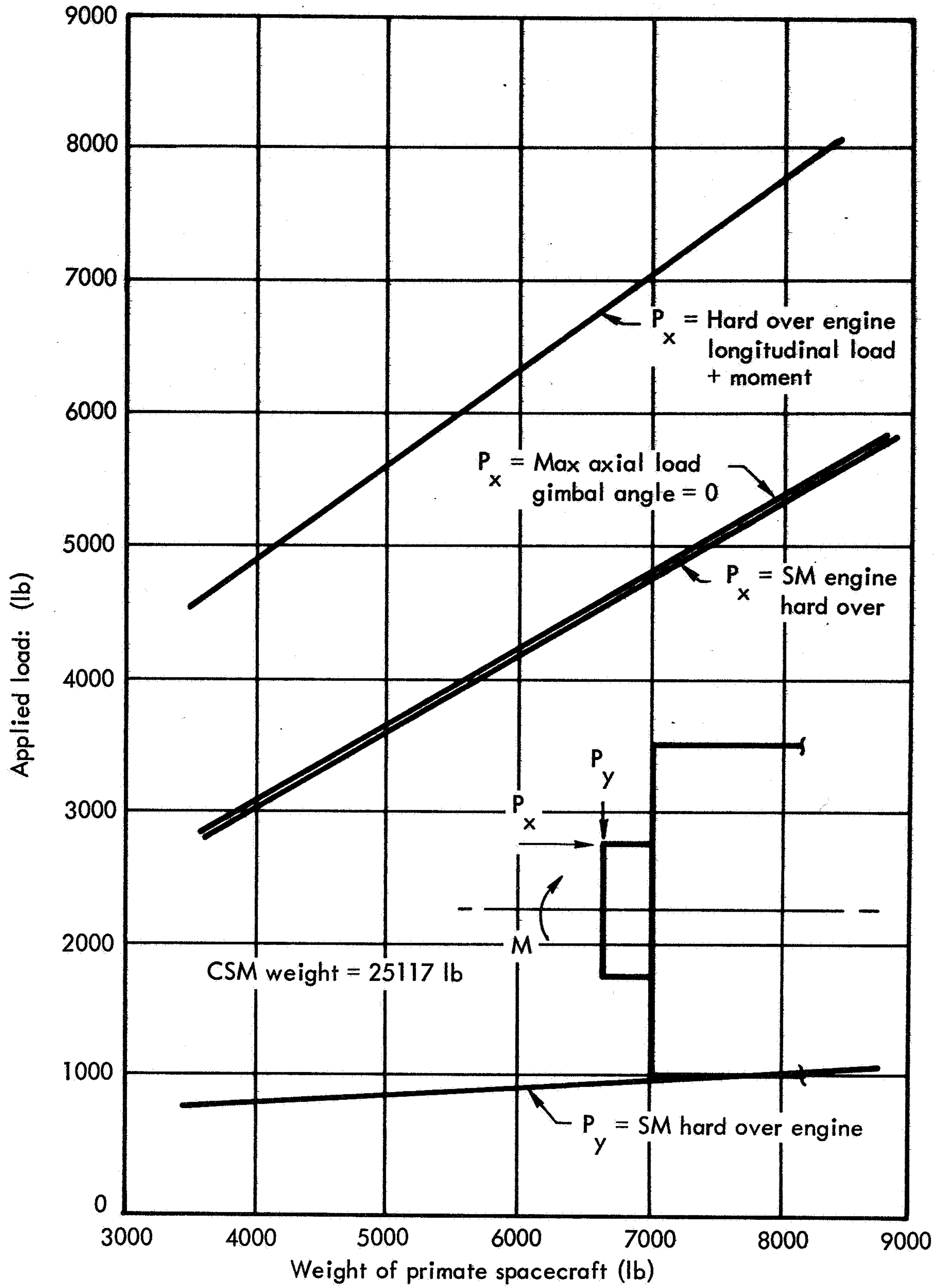


Figure C2. - Injection load

APPENDIX D

TRANSDUCER COMPARISON ANALYSIS

TABLE D1. - TRANSDUCER COMPARISON SUMMARY - RESISTANCE TEMPERATURE PROBE - GAS OR LIQUID

Type transducer							
Item	Comparison parameter	MFdg model ➡	Rosemount 176FL	Nycal RTS-5013-D-LLL	Mincon S31-20A	Thermal systems P300	Transconics T4578-303
1.	Transducer range		0 to 200°F	77°K to 373°K	-148° to 400°F	-430 to 600°F	+40 to +120°F
2.	Parameter sensed		+20 to 120°F	+20 to 120°F	+20 to 120°F	+20 to 120°F	+20 to 120°F
3.	Output voltage		0 - 5.0 vdc, FS	.0 to 50 mv	0 - 50 mv	0 to 50 mv	0 to 50 mv
4.	Output impedance		250 ohms max	800 ± 2 ohms	470 ohms	1400 ohms	1400 ohms
5.	Min external load		25K ohms	800 ohm bridge	470 r bridge	1400 r bridge	1400 r bridge
6.	Output sensitivity				1.0 ohm/°F		1.49 ohm/°C
7.	Responce time		0.5 sec	250 ms	1.3 sec	2.5 sec	1.0 sec
8.	Static/dynamic error		±.2%		*100:10^6	± 0.1%	*±.5°F/±.75°F
9.	Linearity error		±.3% FS		*1%		
10.	Input voltage		28 ± 4 vdc	5 V DC	2 V DC		
11.	Input impedance		source*1.0 ohm				
12.	Max input power		48 ma			15 ma	1 ma
13.	Max overload capability		NA*				
14.	Output resolution		±.3% FS				
15.	Output noise/regulation		*5 mv rms				
16.	Operating temp range		0 to 200°F	10 to 535°K		-420 to +700°F	
17.	Operating vibration		30G peak 5-2 cps			80G,29 - 2M cps	
18.	Max shock		50G ST - 11 ms				
19.	Max acceleration		30G - all axis				
20.	Pressure requirements		sealed case				
21.	Calibration requirements		pin-50±.5% FS		± .25%		
22.	Estimated life - hours		NA*	NA*	NA*	NA*	NA*
23.	Temperature coefficient		.0035% FS/°F		.00392% FS/°C		.003925% FS/°C
24.	Weight		20 oz				2.41 oz
25.	Size, LXWXH		4.075 x 1.75 x 1.75	3.0 x 1" hex	19 x 3/8 hex		3.72 x .852 dia
26.	Unit cost		$1,215.00	$238.00	$91.00	$135.00	$325.00
27.	Space qualified		LEM	Saturn	No	LEM	LEM
28.	Integrated transducer		Yes SC + VC	No	No	No	No

* Not available

TABLE D2. - TRANSDUCER COMPARISON SUMMARY - TEMPERATURE SENSOR

Type transducer		Temperature sensor	
Item	Comparison parameter MFdg model	Microsystems (EOS) type 2002-0034-1	CEC type 4-550-001
1.	Transducer range	-50°F to +150°F	0 to +200°F
2.	Parameter sensed	coolent,+70 to +120°F	coolent, +70 to 120°F
3.	Output voltage	0 - 5 vdc ± 1%	0 to 40 mv
4.	Output impendance	*100 ohms	500 ±25 ohms
5.	Min external load	25K ohms min	
6.	Output sensitivity		4 mv/v
7.	Response time	*1.0 sec	250 msec
8.	Static/dynamic error	±.2% FS	±0.1% FS
9.	Hysteris error	± 1.0%	±.25% FS
10.	Input voltage	28 ± 4 vdc	10 vdc
11.	Input impedance		500 + 25 ohms
12.	Max input power	*50 ma	10 ma
13.	Max overload capability	3000 psi	3000 psi
14.	Output resolution	continuous	continuous
15.	Output noise/regulation	10 mv, OV	
16.	Operating temp range	-100° to 250°F	-100F to +500°F
17.	Operating vibration	25G, 10-2K cps	100G, 5 to 1000 cps
18.	Max shock	60G - 10 msec	
19.	Max acceleration	50G - all axis	
20.	Pressure requirements	sealed	
21.	Calibration requirements	external	internal-pin
22.	Estimated life - hours	NA	NA
23.	Temperature coefficient		
24.	Weight	7 oz	6 oz
25.	Size, LXWXH		4.25 x .687 oz
26.	Unit cost	$952.00	$475.00
27.	Space qualified	Under qual. test	No
28.	Integrated transducer	Yes, SC + VC	No integ. bridge

TABLE D3. - TRANSDUCER COMPARISON SUMMARY - PRESSURE TRANSDUCER GAS 0 - 4000 PSIA

Item	Type transducer Comparison parameter	MFdg model	Micro systems (EOS) 1025 - 0015	Servonic instr. 2091 - 0910	Wianko (Whittaker) P2-3250-3-4000	Trans-sonics P106-3500-2A	BLK electronics DHFM-277822
1.	Transducer range		0 - 5500 psia	0 - 4000 psia	0 - 4000 psia	0 to 3500 psia	0 to 4000 psia
2.	Parameter sensed		GN_2 - 0 to 4000 psia	GN_2 - 0 to 4000 psia	$GN_2$0-4000 psia	$GN_2$0-4000 psia	$GN_2$0-4000 psia
3.	Output voltage		0 to 5 vdc	0 - 5 vdc	0 to 5 vdc	0 - 5 vdc	0 - 40 mv
4.	Output inpedance		100 R- Max	7500 r ± 375 r	200 ohms	7500 R ±5	350 ohms
5.	Min external load				15,000 R, min		
6.	Output sensitivity		± 1.0% FS				
7.	Response time				1.5 msec	*3 msec	*300 N sec
8.	Static/dynamic error		± 2.0% FS	±1.0%/±2.5% FS	±.5%/	±1.5%/.3%	
9.	Hysteris linearity error		± .25% FS		.2%		.5% FS
10.	Input Voltage		20 to 50 vdc	10 VDC	28 vdc ±4	10 vdc	10 VDC
11.	Input impedance						350 ohms
12.	Max input power		5 ma - nom	10 ma - nom	9 ma - nom	10 ma-max	15 ma
13.	Max overload capability		8000 psi			7000 psi	16000 psia
14.	Output resolution		continuous	.25% FS max	continuous	.33%(300 wires)	continuous
15.	Output noise/regulation		5 mv rms		ripple,3% FS OV		
16.	Operating temp range		-100° to +300°F	-100° to +165°F	-65°F to +200°F	-65°F- +200°F	-35°F to +200°F
17.	Operating vibration		10G - 2 to 2Kcps	35 GS max	1% rms - 20G	.3% at 20G	NF = 42 Kc
18.	Max shock		30G for 10 ms			200G-all axis	1000G - 1 msec
19.	Max acceleration		20G - 5 min-all axis	20G - all axis	.05%/G	20G-all axis	15G - all axis
20.	Pressure requirements		sealed	sealed	sealed	sealed	sealed
21.	Calibration requirements		external	external	external	external	internal
22.	Estimated life - hours		500,000 MTBF	NA	NA	100K cycles	
23.	Temperature coefficient		±1.5% 0 to 250°F		±2.5% over range	.01% FS/°F	.005%/°F
24.	Weight		5 oz	5 oz	9 oz	4.5 oz	5 oz
25.	Size, LXWXH		3.25 x 1.50 x 1.0	2.5 x 1.0 dia	3.25 x 1.53 x 1.53	2.72 x 1.0 dia	2.0 x 1-1/8 dia
26.	Unit cost		$1,061.00	$328.00	$1,450.00	$390.00	$375.00
27.	Space qualified		LEM	Saturn	Apollo CM	No	624A (Titan)
28.	Integrated transducer		Yes, SC + VC	No-put type	Yes, SC + VC	No reluct. type	No - S.G. type

TABLE D4. - TRANSDUCER COMPARISON SUMMARY

Item	Comparison parameter (Type transducer / MFdg model)	Wianko (Whittaker) P2-3250-3-50	Trans-sonics P106R-50-2A	Wianko (Whittaker) P2-5101-3-50D	Statham PL749TC-70D	BLH Electronics DHFM - 277775
	Type transducer	Pressure, liquid, 0 - 50 psia & p				
1.	Transducer range	0 - 50 psia	0 - 50 psia	0 - 50 psia	0 - 70 psia	0 - 50 psia
2.	Parameter sensed	H_2O 0 to 50 psi	H_2O 0 - 50 psia	Coolant 0 - 50 psia	Coolant 0-50 psia	Coolant 0-50 psia
3.	Output voltage	0 to 5 vdc	0 to 5 vdc	0 - 5 vdc	0 - 50 mv	0 - 40 mv
4.	Output impedance	200 ohms	7500 ± 5 R	60 ohms	350 ohms	350 ohms
5.	Min external load	15000 R, min		25000 R nom		
6.	Output sensitivity			± .1%	5 mv/v	4 mv/v
7.	Response time	1.5 msec	<3 msec			<500/v sec
8.	Static/dynamic error	±.5%	±1.5% /.3%		.01%	.1%
9.	Hysteris linearity error	.2%		.1%/±.5%	±.75% FS	.25%/.5% FS
10.	Input voltage	28 ± 4 vdc	10 vdc	20 to 36 vdc	5 vdc	10 vdc
11.	Input impedance				350 ohms	350 ohms
12.	Max input range	9 ma	10 ma - max	6 ma	7.2 ma	15 ma
13.	Max overload capability	75 psia	100 psia	100 psia	100 psia	100 psia
14.	Output resolution	Continuous	.33% (300 min)	Continuous	Continuous	Continuous
15.	Output noise/regulation	Ripple 3% FS OV		10 mvpp-max		
16.	Operating temp range	-65° to + 200°F	-65°F to +200°F	-65°F to +200°F	-65 to +400°F	-35°F to +200°F
17.	Operating vibration	1% rms - 20G	.3% at 20G	10 mv - 20G	Nat.Freq.=8.7 Kc	N.F.= 5Kcs - 25G
18.	Max shock		200G - all axis			150G - 5 msec
19.	Max acceleration	.05%/G		.1%/G	.01% FS/G	15G - all axis
20.	Pressure requirements	Sealed	Sealed	Sealed	Sealed	Sealed
21.	Calibration requirements	External-R	External	External	External	Internal
22.	Estimated life - hours	NA	100K cycles			
23.	Temperature coefficient	±2.5% F Range	.007%/°F	±1.0% F. Range	.01%/°F	.01%/°F
24.	Weight	9 oz		7 oz.	7 oz	6 oz.
25.	Size, LXWXH	3.25 x 1.53 x 1.53	3.75 x 1.25 x 1.4	2.5 x 1.5 x 1.0	.258 x 1.12 dia	3.1 x 1.0 dia
26.	Unit cost	$1,450.00	$390.00	$1,800.00	$405.00	$425.00
27.	Space qualified	Apollo CM	NO	LEM	Yes - Qualified	Titan
28.	Integrated transducer	Yes - SC + VC	NO - pot type	Yes - SC + VC	No - SG type	No - SG type

TABLE D5. - TRANSDUCER COMPARISON SUMMARY - LIQUID FLOW METER

Type transducer			
Item	Comparison parameter / MFdg model	Rosemount 124AM227G1	Microdot R-3/16 - 3J
1.	Transducer range	0 to 22.7 cc/min	0 to 1.G/min
2.	Parameter sensed	H_2O 0 - 31.5 cc/min	H_2O 0 - 31.5 cc/min
3.	Output voltage	0 to 5 vdc	0 to 30 mvpp
4.	Output impedance	< 500 ohms	900 ohms
5.	Min external load		
6.	Output sensitivity	4.5 cc/min/V_o	126 cc/min/mv
7.	Response time	.3 sec	10 msec
8.	Static/dynamic error		0.2%
9.	Hysteris error	±.5% FS OV	0.5% FS OV
10.	Input voltage	28 ± 4 vdc	5VDC
11.	Input impedance		
12.	Max input power	3.5 watts	10 ma
13.	Max overload capability	1000 psi	150 psi
14.	Output resolution	continuous	continuous
15.	Output noise/regulation		
16.	Operating temp range	+50 to 90° F	-320°F to +350°F
17.	Operating vibration	5G-20 to 2Kcps	
18.	Max shock	30G - 8 msec	
19.	Max acceleration	10G - all axis	
20.	Pressure requirements	sealed	
21.	Calibration requirements	oper. check pin	
22.	Estimated life-hours	NA	
23.	Temperature coefficient	±.1% FS from 70°F	
24.	Weight	40 oz	
25.	Size, LXWXH	4.6 x 3.33 x 3.5	2.219 x .75 x 2.9
26.	Unit cost	$3,310.00	$450.00
27.	Space qualified	LEM	no
28.	Integrated transducer	yes - SC + VC	no - coil type

TABLE D6. - TRANSDUCER COMPARISON SUMMARY - ACCELEROMETER, VIBRATION MEASURING

Item	Comparison parameter MFdg model	Endevco-accel 2260	CEC-accel 4 - 202 - 0001	CEC-vibration 4 - 123A
1	Transducer range	0 to 2000 cps	200 to 2500 cps	45 to 2000 cps
2	Parameter sensed	20 cps to 2Kcps	20 cps to 2Kcps	20 cps to 2Kcps
3	Output voltage	± 250 mv	0 to 40 mv	
4	Output impedance	500 ohms	350 ohms	
5	Min external load	100K ohms	100 K ohms	
6	Output sensitivity	25 mv/v	8 mv/v	135 mv/in/sec
7	Response time	.01 of critical D	.7 of critical D	
8	Static/dynamic error	±7 mv	±5%	
9	Hysteris error	±1%	±.75%	
10	Input voltage	10 vdc	5VDC	
11	Input impedance	500 ohms	350 ohms	
12	Max input power			
13	Max overload capability		750G	
14	Output resolution	continuous	continuous	
15	Output noise/regulation			
16	Operating temp range	-65°F to +250°F	-65°F to +250°F	-65° to +500°F
17	Operating vibration	N.F. = 12 Kc ±750G	N.F. = 3Kc ±500G	
18	Max shock	±750G	100G - 11 msec	
19	Max acceleration	±750G	+500G	
20	Pressure requirements	sealed	sealed	
21	Calibration requirements	external	external	
22	Estimated life - hours	NA	NA	NA
23	Temperature coefficient	.02%/°F	.01%/°F	
24	Weight	1.25 oz	3 oz	4.25 oz
25	Size LXWXH	5/8 hex x 1.2	1 x 1 x 1.04	
26	Unit cost	$350.00	$300.00	$195.00
27	Space qualified	no	no	no
28	Integrated transducer	no - SG type	no - SG type	no - SG type

TABLE D7. - TRANSDUCER COMPARISON SUMMARY - EVENTS, SWITCHES AND VOLTAGE SENSORS

Type transducer			
Item	Comparison parameter MFdg model	Microswitch (honeywell) 2HS	Bourns 3910
1.	Transducer range	SPDT	DPDT
2.	Parameter sensed	Event closure	Event closure
3.	Output voltage	5 vdc	5 vdc
4.	Output impedance	1 ohm	.2 ohms
5.	Min external load	1 amp	1 amp
6.	Output sensitivity	---	---
7.	Response time	2 msec	10 msec
8.	Static/dynamic error	---	±5%
9.	Hysteris error	---	±1%
10.	Input voltage	5 vdc	-24 to +6 vdc, +6 to +12 vdc, +12 to 100 vdc
11.	Input impedance	1 ohm	2000 r/v
12.	Max input power	Oper. Force=7 oz max	Trigger over L = 400%
13.	Max overload capability	---	
14.	Output resolution	---	Trigger dropout = -4%
15.	Output noise/regulation	---	Supply Voltage = 20-30V
16.	Operating temp range		-55 to +105°C
17.	Operating vibration		20G 10 to 2Kcps
18.	Max shock		75G - 11 msec
19.	Max acceleration		75G - all axis
20.	Pressure requirements	Sealed	Sealed
21.	Calibration requirements	None	None
22.	Estimated life - hours	NA	100K cycles
23.	Temperature coefficient	---	---
24.	Weight	.42 oz	1.0 oz
25.	Size, LXWXH	1.79" x .25" x 1.18"	1.0 x 1.0 x .51
26.	Unit cost	$11.00	$77.00
27.	Space qualified	No	No
28.	Integrated transducer	No - SPDT HS	No - DPDT

TABLE D8. - TRANSDUCER COMPARISON SUMMARY - VOLTAGE PICKOFFS AND CURRENT PICKOFFS, NORTHROP SYSTEMS LABORATORIES

Type transducer		
Item	Comparison parameter MFdg model	
1.	Transducer range	All voltage and current pickoffs shall be an integral part of the specific subsystem electronics involved. (1) Standard electronic practice of the use of current limiter resistors. EMI shunt capacitors, and voltage limiting diodes shall be incorporated for subsystem electronic protection. (2) Current pickoffs shall utilize standard shunt millivolt resistances equipped with current limiters for circuit protection.
2.	Parameter sensed	
3.	Output voltage	
4.	Output impedance	
5.	Min external load	
6.	Output sensitivity	
7.	Response time	
8.	Static/dynamic error	
9.	Hysteris error	
10.	Input voltage	
11.	Input impedance	
12.	Max input power	
13.	Max overload capability	
14.	Output resolution	
15.	Output noise/regulation	
16.	Operating temp range	
17.	Operating vibration	
18.	Max shock	
19.	Max acceleration	
20.	Pressure requirements	
21.	Calibration requirements	
22.	Estimated life- hours	
23.	Temperature coefficient	
24.	Weight	
25.	Size, LXWXH	
26.	Unit cost	
27.	Space qualified	
28.	Integrated transducer	

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